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ADVANCED COOLED TURBINE AIRFOIL AERODYNAMIC INVESTIGATION

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NOTICE

This technical report has been reviewed and is approved for publication.

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Several convectively cooled trailing edge designs were investigated with the objective of eliminating the need for film cooling on the airfoil suction side. A 43.4% reduced solidity first stage turbine vane having potential application for an advanced Air Force fighter engine was selected for the evaluation. Primary cooling candidates were: cross flow/impingement and wavy criss-cross slot. The final design eliminates film cooling on the suction side and uses the wavy criss-cross slot as the cooling scheme for the trailing edge section.

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The cooling design was incorporated into a cascade test airfoil using the radial wafer fabrication technique. The airfoil was constructed by photoetching the cooling design into the individual wafers, bonding the wafers together and machining the bonded block into the airfoil shape. The airfoil was subsequently evaluated in an airfoil cascade test to determine the aerodynamic and cooling performance. The aerodynamic profile loss of the reduced solidity radial wafer airfoil was reduced 56% relative to a baseline 43.4% reduced solidity configuration with film cooled suction surface and was 30% under the program goal. The heat transfer data showed a 10% improvement in hot spot cooling effectiveness compared to conventional pedestal designs, and the suction side trailing edge temperatures were approximately 100°F cooler than predicted. The wavy criss-cross slot design used in the trailing edge section proved to be an efficient cooling technique, and eliminated the need for suction side film cooling.

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PREFACE

This final report was submitted by Pratt & Whitney Aircrat Group Government Products Division under Contract F33615-76-C-2009. The effort was sponsored by the Air Force Aero Propulsion Laboratory, Air Force Systems Command, Wright-Patterson AFB, Ohio under Project No. 3066, Task No. 06 and Work Unit No. 25 with Wayne A. Tall as Project Engineer. W. G. Hess of Pratt & Whitney Aircraft Group Government Products Division was technically responsible for the work.

TABLE OF CONTENTS

LIST OF ILLUSTRATIONS

LIST OF ILLUSTRATIONS (Continued)

LIST OF TABLES

SECTION I

INTRODUCTION AND SUMMARY

A. INTRODUCTION

Cooling studies of advanced reduced solidity turbines indicate that in order to meet the increased heat loads, conventionally cooled trailing edge designs must be augmented by film cooling to effectively lower the mainstream gas temperature adjacent to the walls. Unfortunately, use of film cooling, especially in the high Mach number regions of the airfoil, increases aerodynamic losses which are proportional to the Mach number squared. The high Mach numbers occur on the suction side of the airfoil in the region of the gage point downstream to the trailing edge. The trailing edge region of turbine airfoils is difficult to cool because (1) the cooling air is hot when it arrives at the trailing edge since it has been used to cool other portions of the airfoil, and (2) the trailing edge thickness is relatively thin, and, in the past, has restricted convective cooling geometries to drilled passages or cast pedestals which are amenable to fabrication by conventional manufacturing methods. This problem is aggravated in reduced solidity designs with increased suction side Mach number and longer surface distance from gage point to trailing edge.

The objective of this program, titled "Advanced Cooled Turbine Airfoil Aerodynamic Investigation", was to demonstrate an improved low solidity turbine airfoil design for advanced high temperature turbines. Specifically, the program was aimed at demonstrating an airfoil design with improved convective cooling features in the trailing edge that minimize or negate film cooling on the suction side which is detrimental to aerodynamic performance. The aerodynamic performance goal of the airfoil design, with 43% reduced solidity, was set equal to that obtained with the conventional solidity airfoil. The cooling goal was to obtain an airfoil that operated with a maximum wall temperature below the life requirement limit of 2000°F.

The program was divided into three separate tasks: Task I consisted of studying various improved convective cooling schemes for the trailing edge which eliminated the need for film cooling on the airfoil suction side; in Task II, the best trailing edge cooling scheme identified in Task I was incorporated into a test airfoil which was representative of an advanced reduced solidity turbine vane mid-section having potential application for an advanced Air Force fighter engine; and subsequently, in Task III, a cascade test was conducted at reduced temperature and pressure to determine the aerodynamic and cooling performance of the cooling design.

B. SUMMARY

Several convectively cooled trailing edge designs were investigated with the objective of eliminating the need for film cooling on the airfoil suction side. A 43.4% reduced solidity first-stage turbine vane having potential application for an advanced Air Force fighter engine was selected for the evaluation. Primary cooling candidates were: cross flow/impingement and wavy crisscross slot. The final design, shown in Figure 1, eliminates film cooling on the suction side and uses the wavy crisscross slot as the cooling scheme for the trailing edge section.

The cooling design was incorporated into a cascade test airfoil using the radial wafer fabrication technique. The airfoil was constructed by photoetching the cooling design into the individual wafers, bonding the wafers together and machining the bonded block into the airfoil shape. The airfoil was subsequently evaluated in an airfoil cascade test to determine the aerodynamic and cooling performance. As shown in Figure 2, the aerodynamic profile loss of the reduced solidity radial wafer airfoil was reduced 56% relative to a baseline 43.4% reduced solidity configuration with film cooled suction surface and was 30% under the program goal. The heat transfer data showed a 10% improvement in hot spot cooling effectiveness compared to conventional pedestal designs, and the suction side trailing edge temperatures were approximately 100°F cooler than predicted. The wavy crisscross slot design used in the trailing edge section proved to be an efficient cooling technique, and eliminated the need for suction side film cooling.

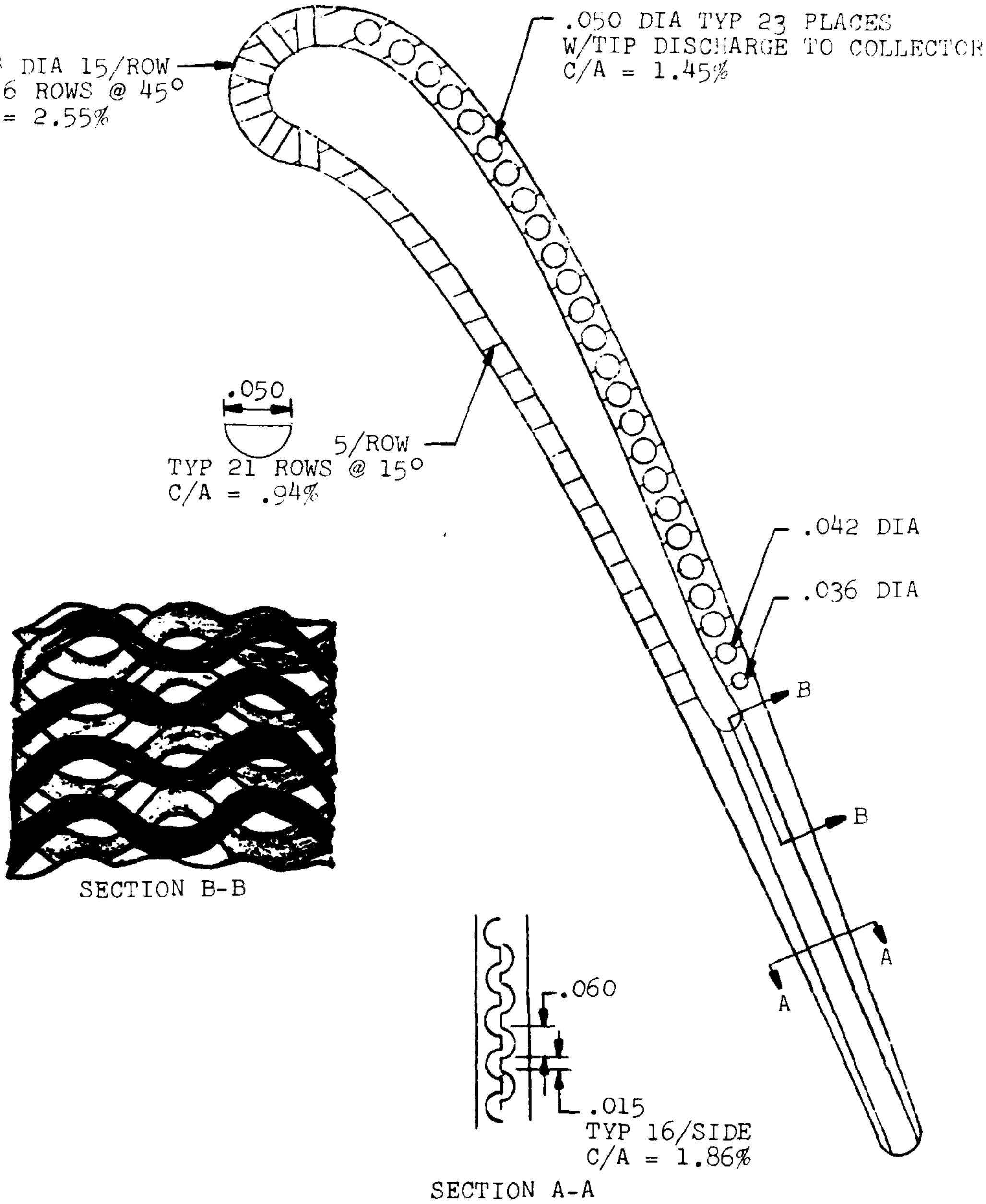

Figure 1. 43.4% Reduced Solidity Radial Wafer Airfoil - Final Design

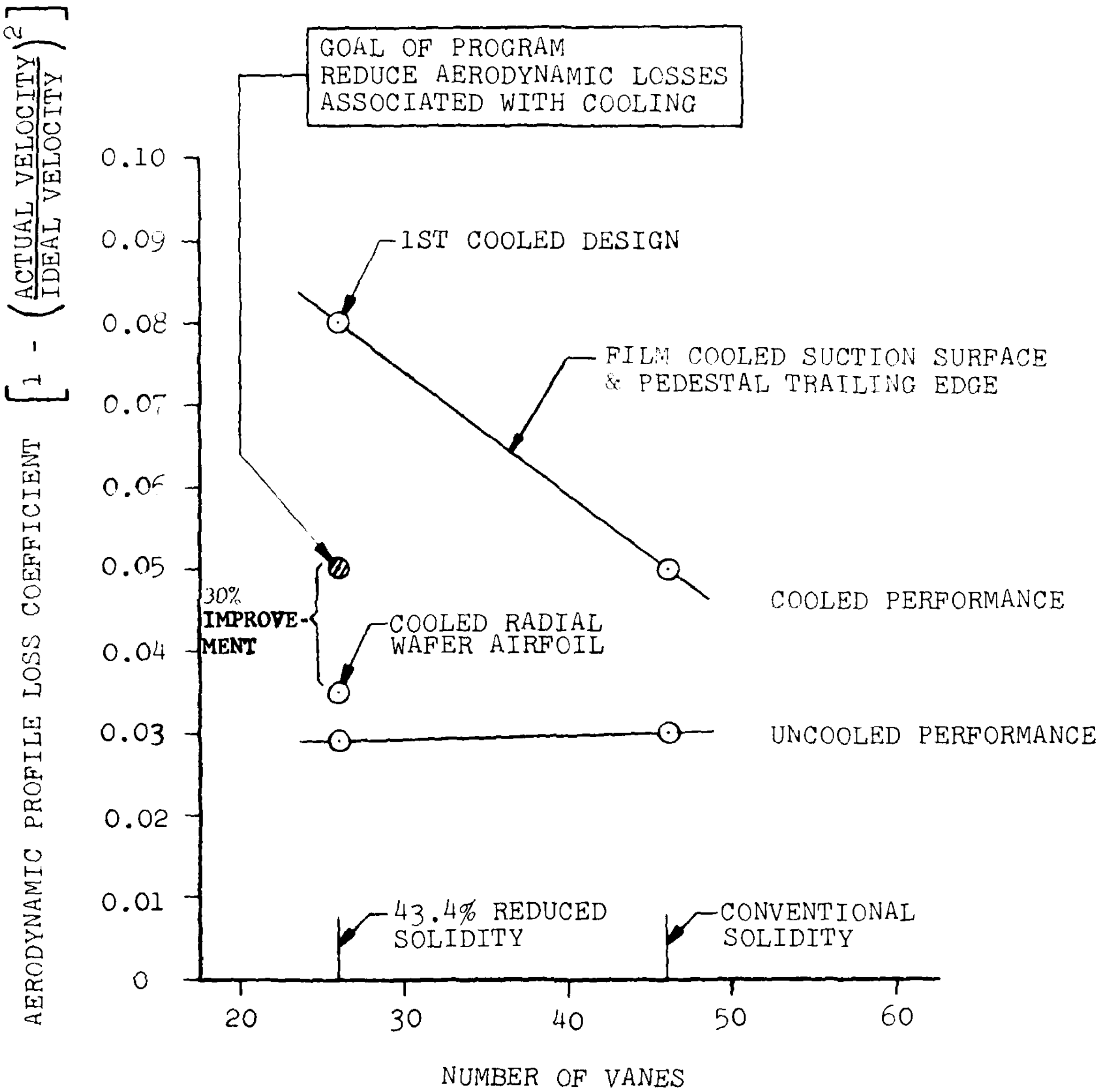

Figure 2. Aerodynamic Performance for First Vane Airfoils

SECTION II

CONCLUSIONS AND RECOMMENDATIONS

A. CONCLUSIONS

Use of an efficient convective cooling scheme on the suction side and in the trailing edge region of an advanced first vane airfoil eliminated the need for film cooling in those regions.

Elimination of the suction side film cooling resulted in an airfoil design with a reduction in the aerodynamic loss of approximately 30% less than the design goal which was set equal to that obtained with the cooled conventional solidity airfoil.

An effective convective cooling scheme was possible through the use of the radial wafer fabrication techniques which permit designs with small intricate convective passages not attainable in cast and drilled airfoils.

The wavy crisscross trailing edge design required a relatively high pressure ratio which makes it primarily suited for the latter rows of a two-stage turbine or the first row of a single stage turbine. The cooling design is also compatible with either the radial wafer or two-piece airfoil fabrication technique.

B. RECOMMENDATIONS

It is recommended that variations in the wavy crisscross slot design, such as number of turns, size of passage, etc., be investigated to obtain a correlation to optimize the cooling effectiveness of the scheme and reduce the pressure drop requirement.

A sensitivity study should be conducted to define the solidity limit such that further reductions in solidity would result in a decrease in aerodynamic performance.

It is further recommended that a program be sponsored to build and test simulated engine turbine hardware for structural integrity and conduct a life cycle cost study to evaluate the advantages of the optimum cooling design in reduced solidity turbine airfoils for future Air Force fighter engines. Demonstration testing of radial wafer airfoils in a selected high temperature gas generator or core engine would be a logical conclusion to the program.

Efforts to reduce the fabrication costs of the radial wafer airfoils should be continued. One such in-house study is being investigated in conjunction with automated casting techniques.

SECTION III

AIRFOIL DESIGN STUDIES

The pressure profile and cross section for the 43.4% reduced solidity airfoil used in this program is presented in Figure 3. The corresponding data for the conventional solidity vane is included for comparison. The pressure profile data illustrate the low suction side static pressures, i.e., high Mach numbers, associated with the reduced solidity design and the need to eliminate or minimize film cooling in that area.

Detailed heat transfer investigations were conducted on the two candidate trailing edge cooling designs, crossflow/impingement and the wavy crisscross slot presented in Figure 4, and on several modifications of each. The analysis included determining the external heat transfer coefficients; gas stream film temperatures; coolant flow distribution, heat transfer coefficients, and pressure losses; and airfoil wall temperatures. The airfoil design conditions are listed in Table 1. For the design phase, a hot spot temperature limit of 2000°F was determined as the maximum allowable for the airfoil to meet the life requirements.

The final crossflow/impingement design investigated is shown in Figure 5. For this design the number of impingement holes in the beginning of the trailing edge section and the discharge gap at the end of the trailing edge section were both increased relative to the initial design to increase the coolant flow through the trailing edge. However, even with these modifications, the predicted wall temperature for the trailing edge section exceeded the life requirement limit as shown in Figure 6.

The wavy crisscross slot design for the trailing edge section was investigated with and without a splitter plate. The purpose of the splitter plate, shown in Figure 7, was to increase the coolant surface area and ultimately lower the wall temperature. After several design analyses, however, it was decided to remove the splitter plate because it eliminated the turbulence caused by the two coolant paths crossing one another.

The final design shown in Figure 8 consists of a six-row showerhead, film cooling on the pressure side, radial cooling passages on the suction side, and the wavy crisscross slot in the trailing edge. The coolant in the radial passages discharges into the vane platform area and ultimately can be used to cool the blade rub strip or the platform since the first vane coolant can be supplied from either OD or ID. The double use of this coolant is a potential means to increase the engine cycle efficiency by reducing the total amount of coolant required. The predicted metal temperature profile for the final design is shown in Figure 9. As indicated, the suction side trailing edge metal temperature is below the life requirement limit of 2000°F.

Table 1. Airfoil Design Parameters for the 43.4% Reduced Solidity Radial Wafer Airfoil

Turbine Inlet Temperature	2660°F
Burner Pattern Factor	0.35
Turbine Inlet Pressure	341 psia
Engine Flowrate	146 lb/sec
Number of Vanes	26
Coolant Temperature	1007°F
Coolant Supply Pressure	348 psia

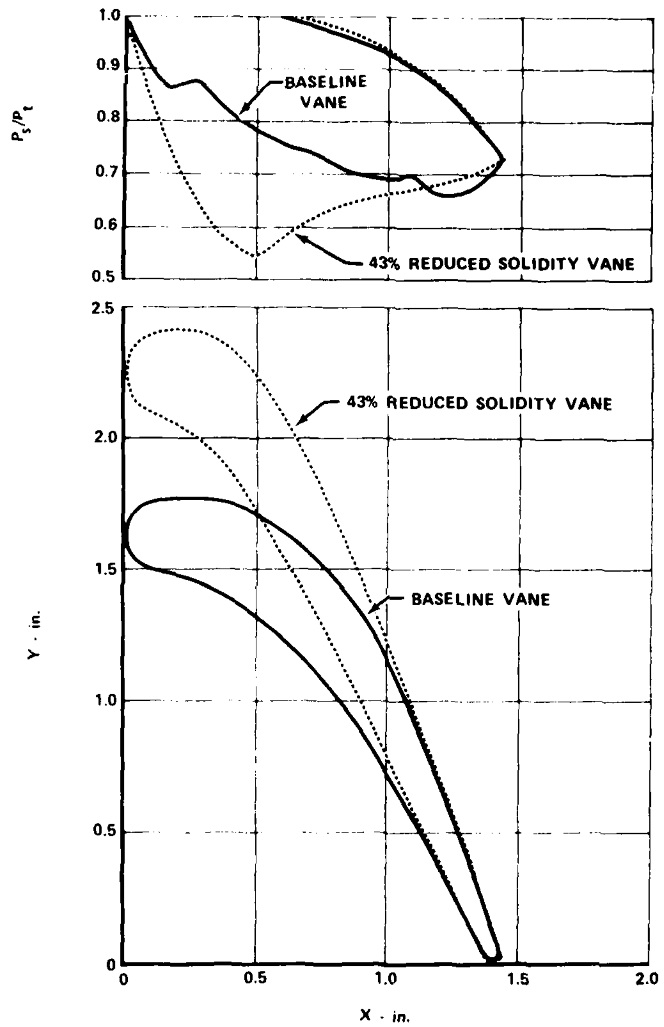

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Figure 3. Pressure Profiles for Conventional and 43% Reduced Solidity Vanes

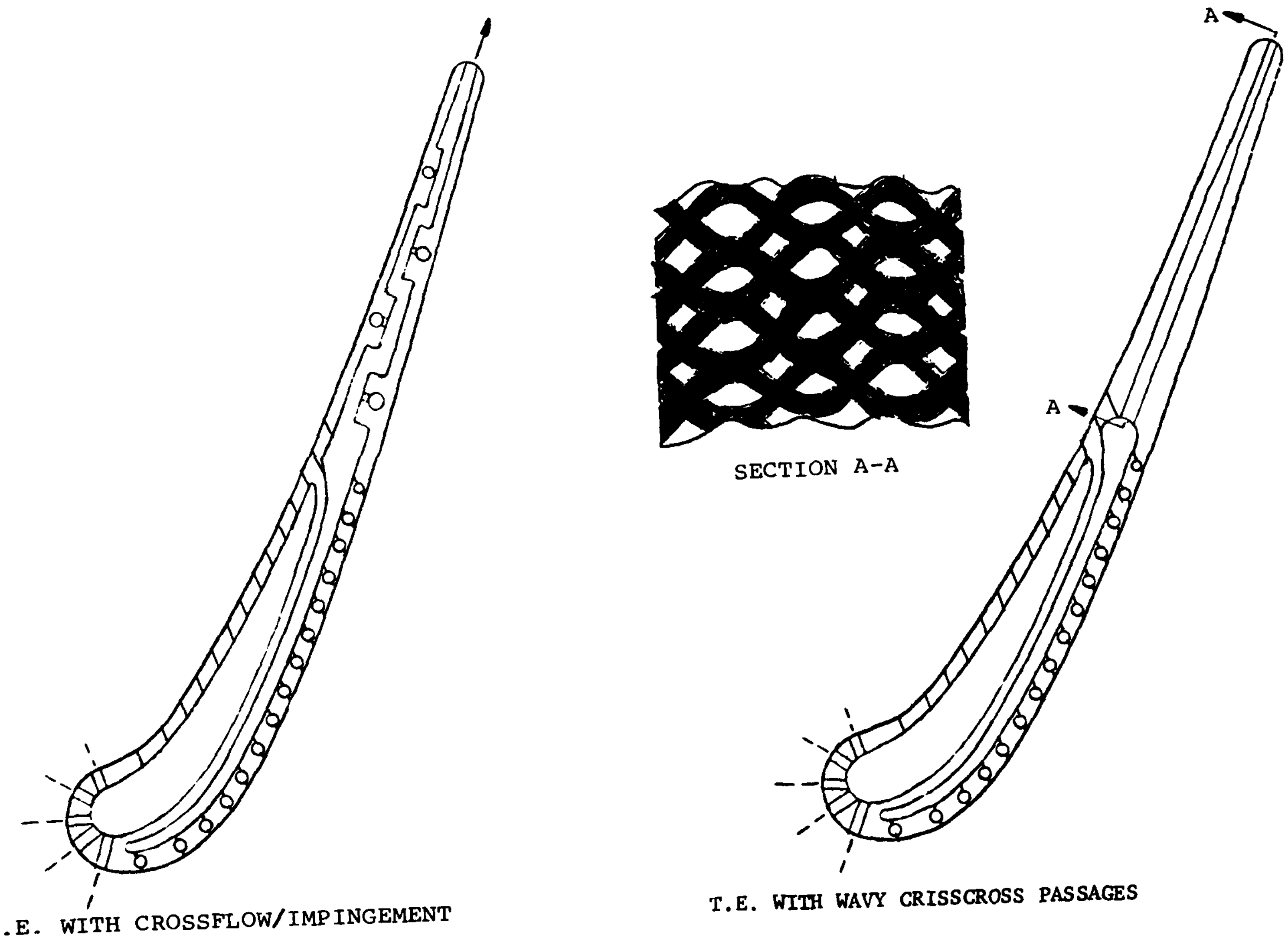

Figure 4. Advanced Cooled Turbine Airfoil Designs

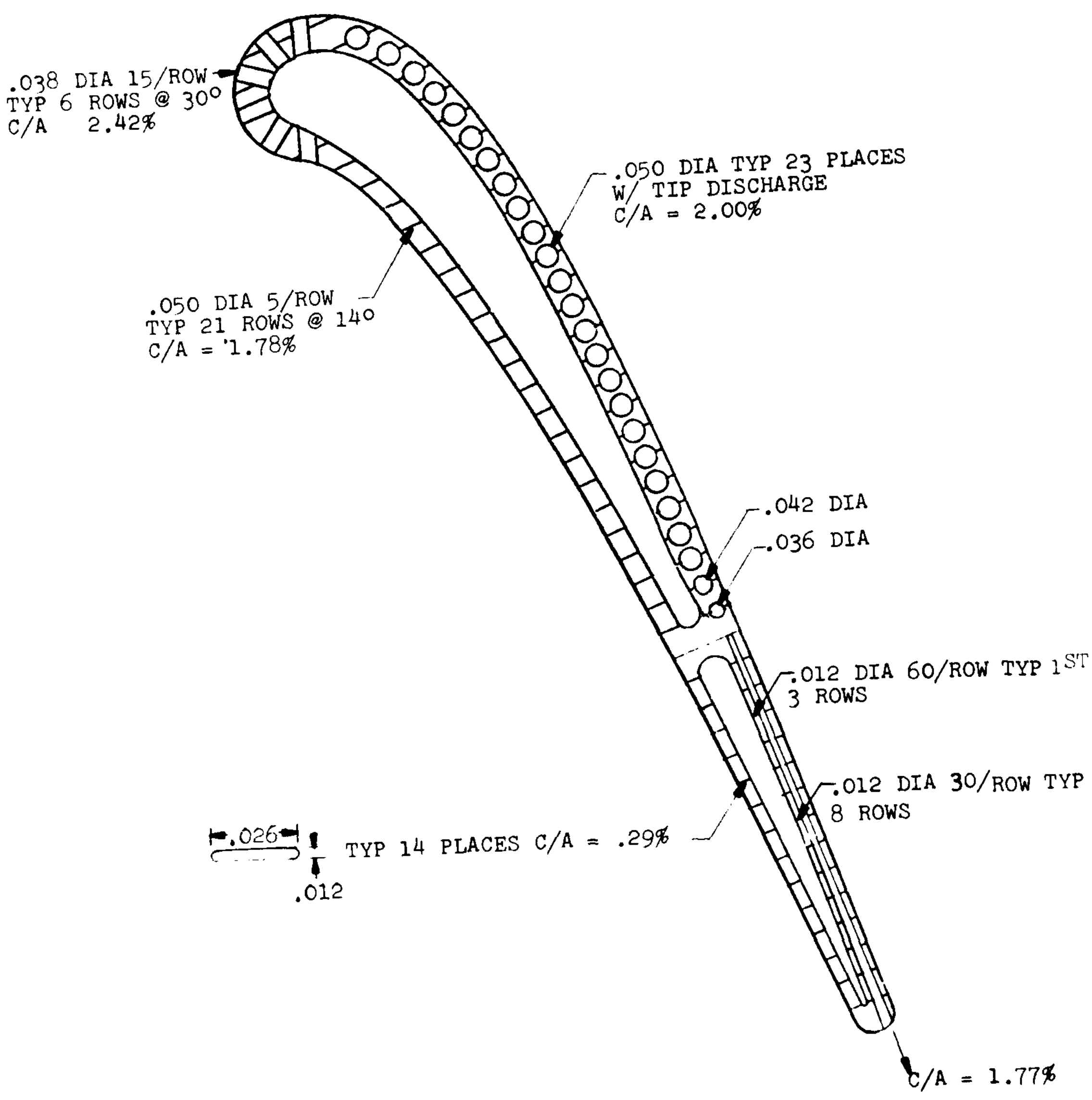

Figure 5. 43.4% Reduced Solidity Crossflow/Impingement Design

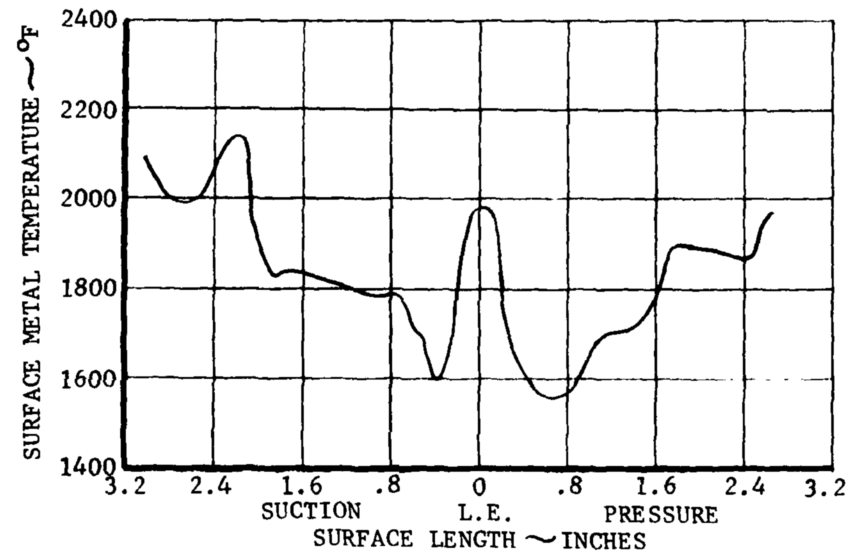

Figure 6. Metal Temperature Profile for Crossflow/Impingement Design

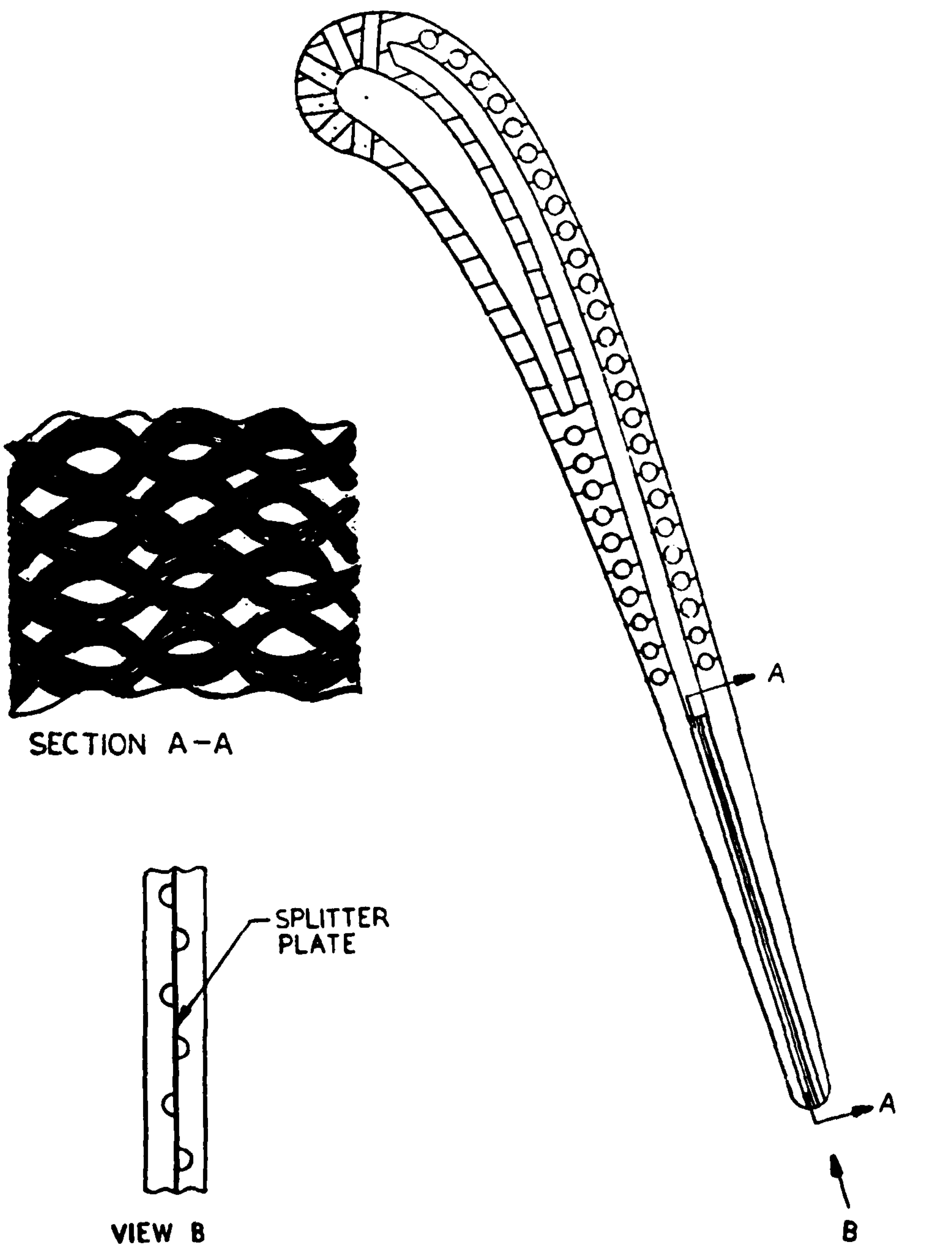

Figure 7. Advanced Cooled Turbine Vane - Wavy Crisscross Trailing Edge Configuration with Splitter Plate

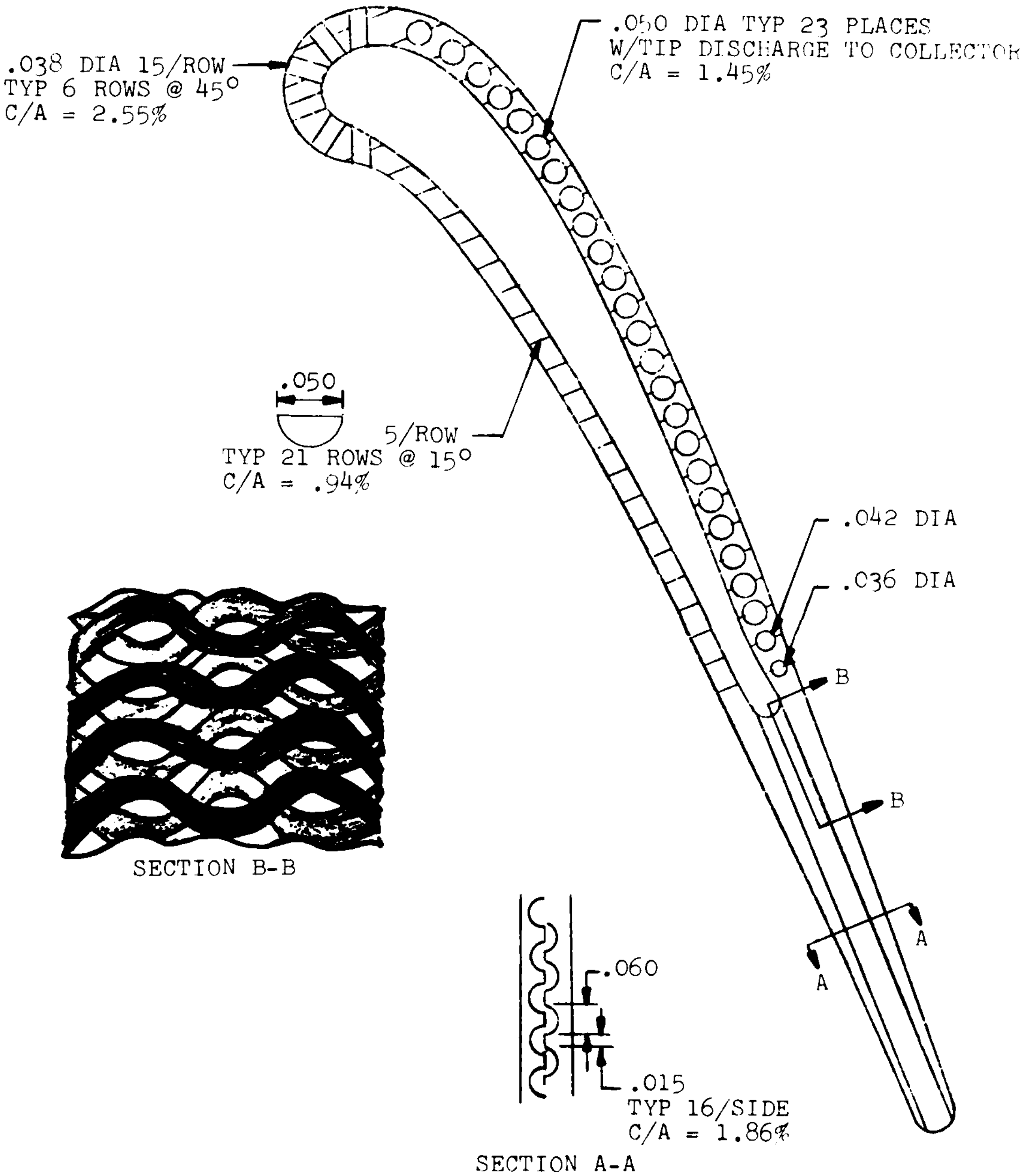

Figure 8. 43.4% Reduced Solidity Radial Wafer Airfoil - Final Design

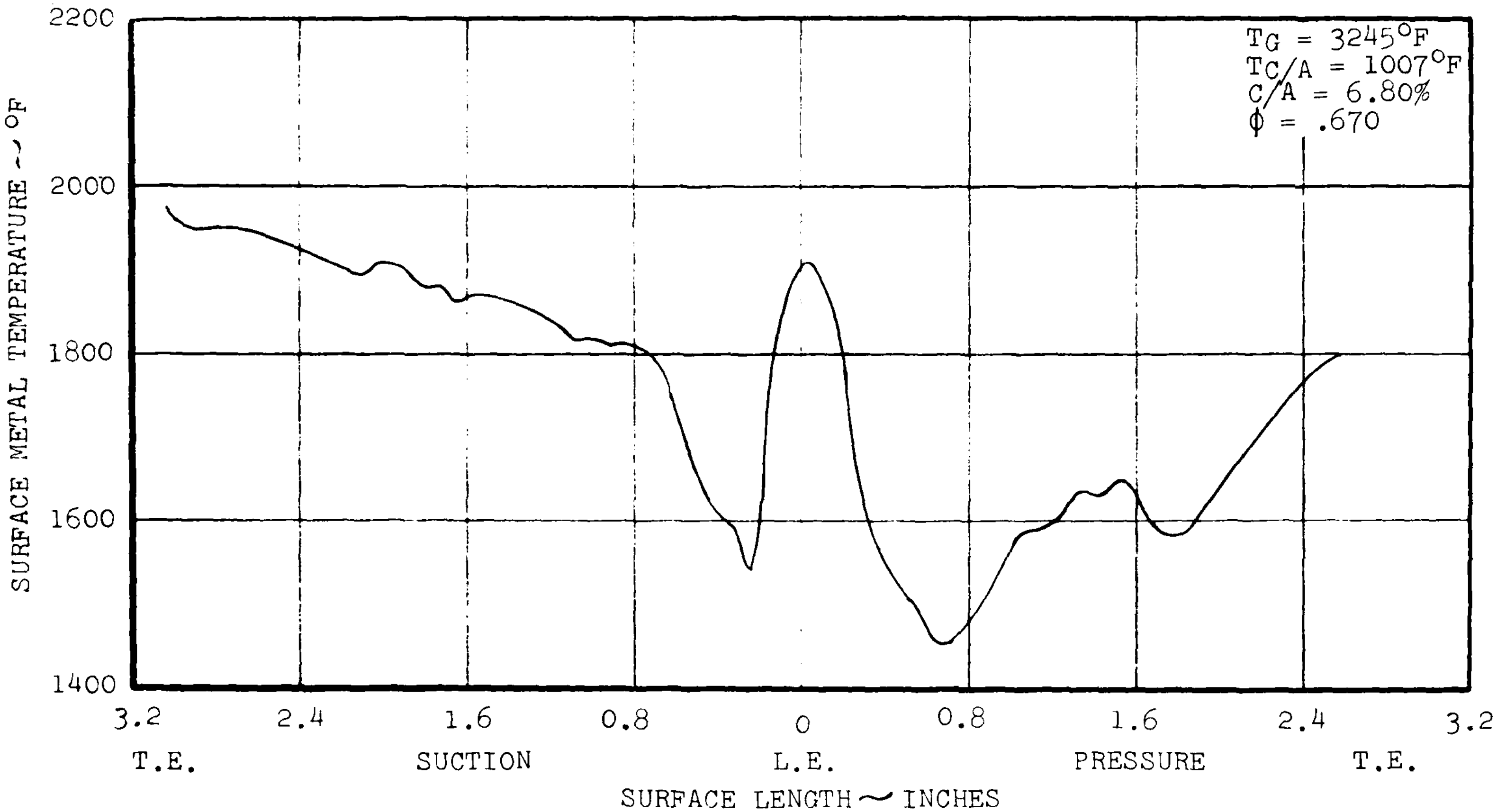

Figure 9. Predicted Metal Temperature Profile for the 43.4% Reduced Solidity Radial Wafer Airfoil Final Design

SECTION IV

AIRFOIL HARDWARE DESIGN AND FABRICATION

The cooling design was incorporated into a cascade test airfoil using the radial wafer fabrication technique. The airfoil is a constant cross-section airfoil with a 3.0 in. span. The wafers which formed the trailing edge cooling design of a wavy crisscross slot were fabricated first since they were bonded in a subassembly and then bonded with the remaining wafers to form the block from which the cascade airfoil was machined. A photograph of the wafers is presented in Figure 10. The wafers were bonded together using a joining process called Transient Liquid Phase (TLP™)[1] bonding. A photograph of the trailing edge block is shown in Figure 11.

The cascade airfoil consists of 27 individual wafers and the trailing edge block as shown in Figure 12. During the etching of the 27 wafers which form the main body of the airfoil, problems occurred and the first set of wafers were rejected due to unacceptable passage size tolerance variations, passage channel irregularities and the resist coating (protective layer which prohibits etching) breakdown. A photograph of one of the rejected wafers (number 18) is presented in Figure 13 showing the resist coating breakdown and passage irregularity. A study of the various etching problem areas, such as resist solution, pattern printing, etching solution mixture, and etching nozzle pressure was conducted and upon obtaining satisfactory results, a second set of wafers were etched which were dimensionally correct and were accepted. Figure 14 illustrates the improvement in the quality of the second etching process by presenting the second etched wafer number 18.

The wafers and the trailing edge block were then TLP bonded together; a photograph of the block which formed the cascade airfoil is shown in Figure 15. The cascade airfoil shown in Figure 16 was then obtained by machining the internal cavity, external contour, and showerhead holes by electrical discharge machining (EDM). The final steps in the fabrication process were the machining of the thermocouple instrumentation slots and the addition of the coolant manifold and supply tubes. A photograph of the completed airfoil with the thermocouple instrumentation slots and the supply manifolds is presented in Figure 17.

1. TLP bonding is a Pratt & Whitney Aircraft process for joining superalloys. The process achieves near parent metal strength but requires only moderate bonding pressures of 15 to 20 psi, compared to conventional diffusion bonding pressures, which are generally in excess of 2000 psi.

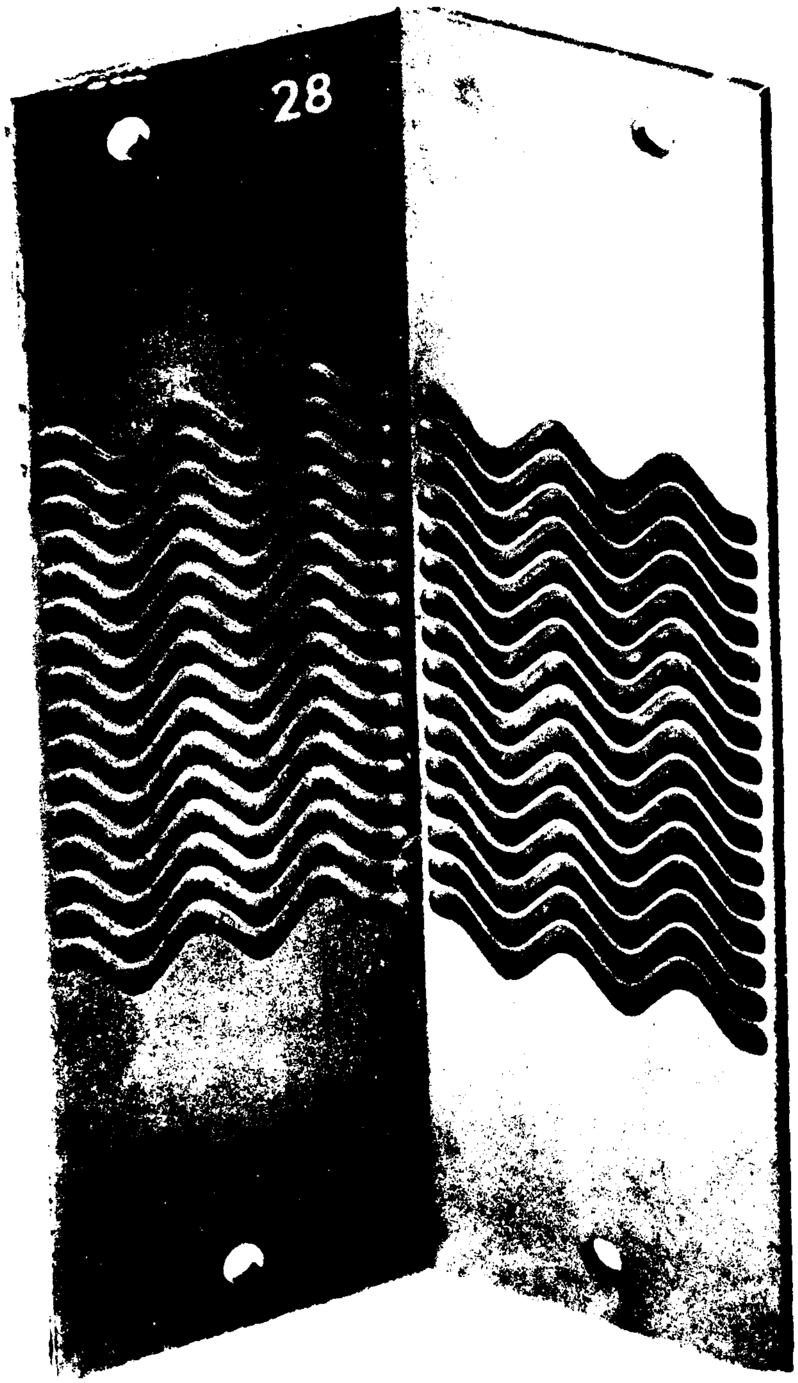

Figure 10. Trailing Edge Wafers for 43.4% Reduced Solidity First Vane Airfoil

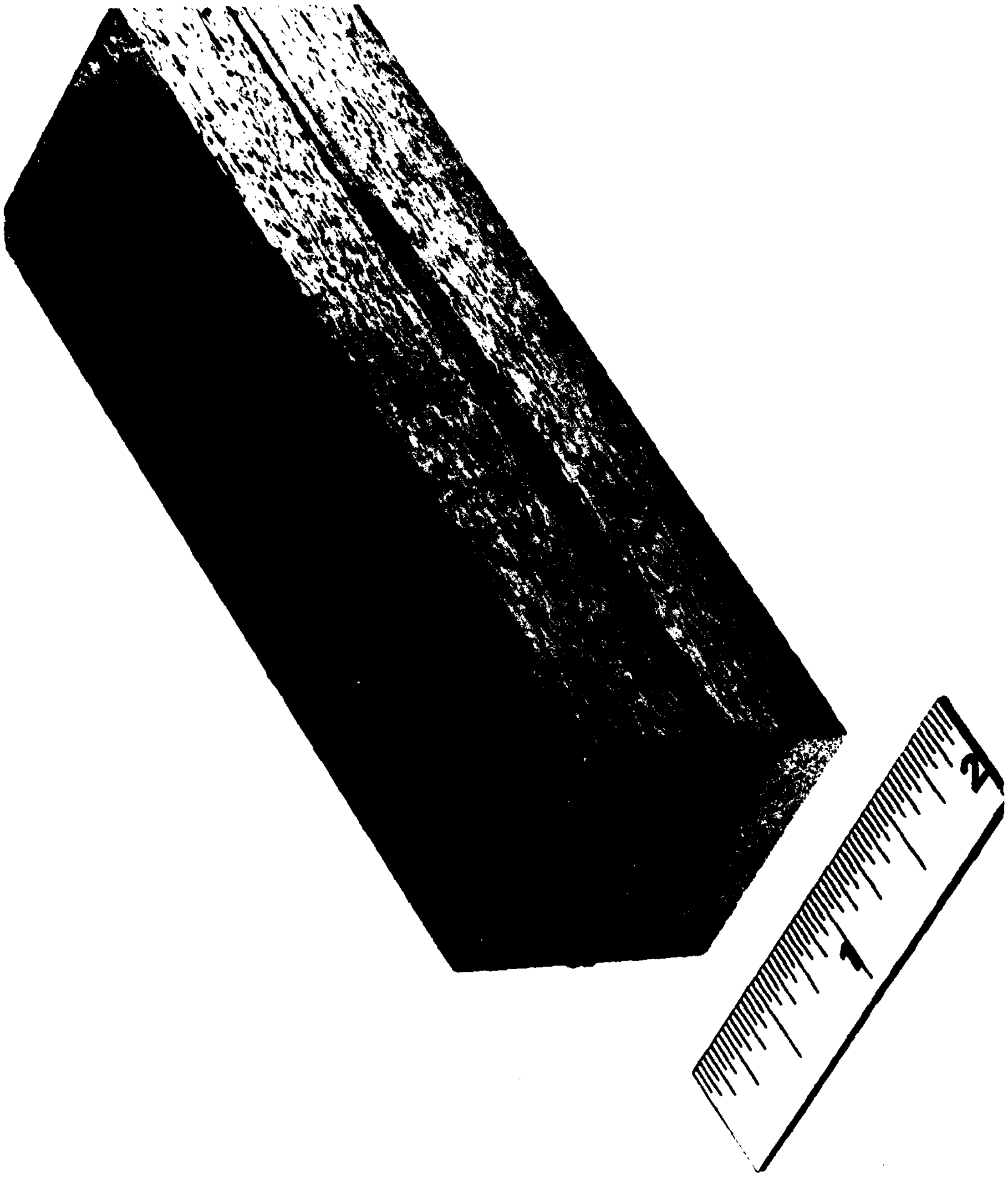

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Figure 11. Trailing Edge Block for the 43.4% Reduced Solidity Radial Wafer Airfoil

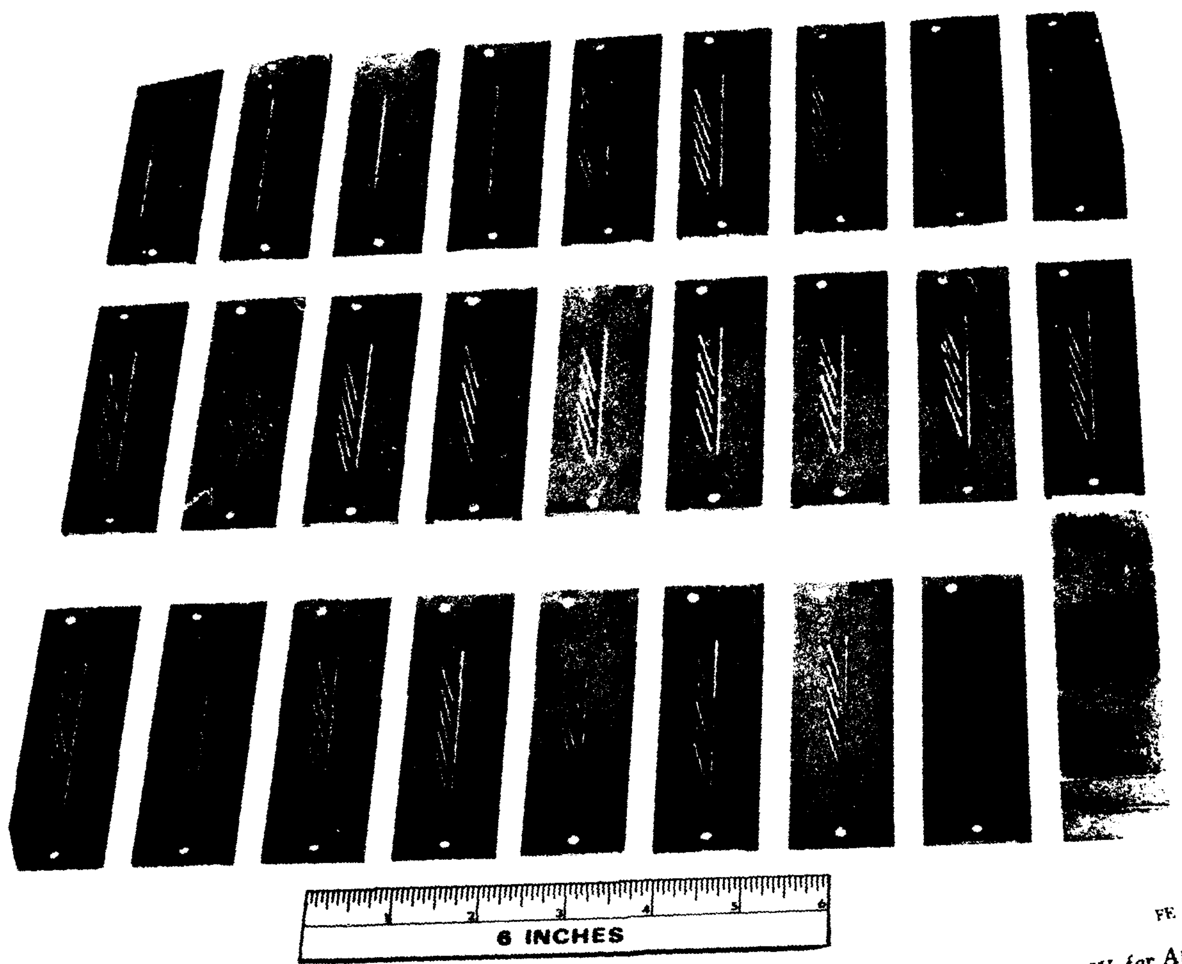

Figure 12. First Set of Etched Wafers and Trailing Edge Block for the 43.4% Reduced Solidity Radial Wafer Airfoil

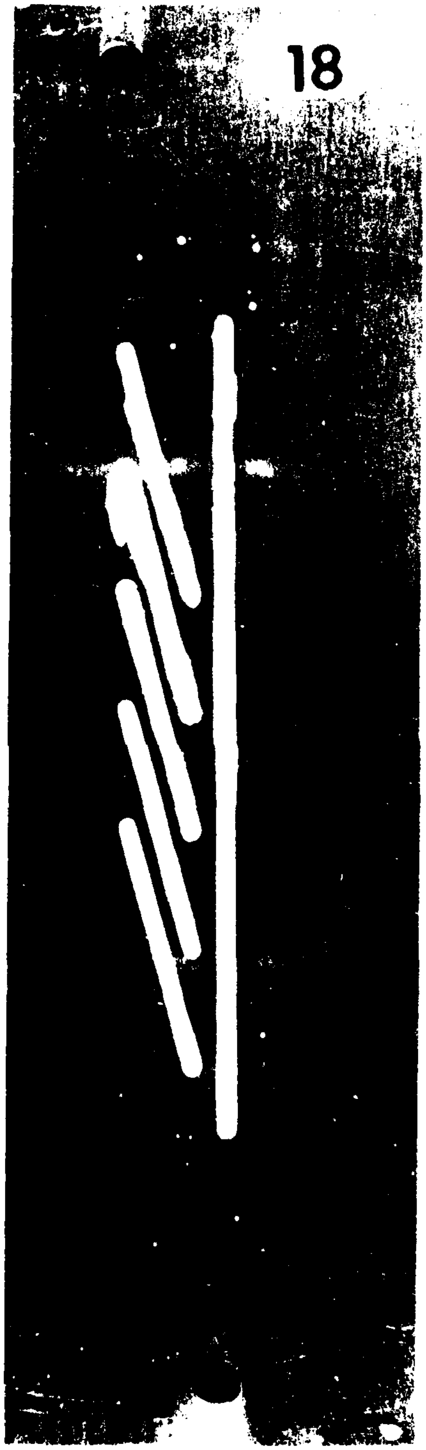

Figure 13. Wafer Etching Showing Resist Breakdown and Passage Irregularities

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Figure 14. Wafer Etching Showing Improved Passage Definition

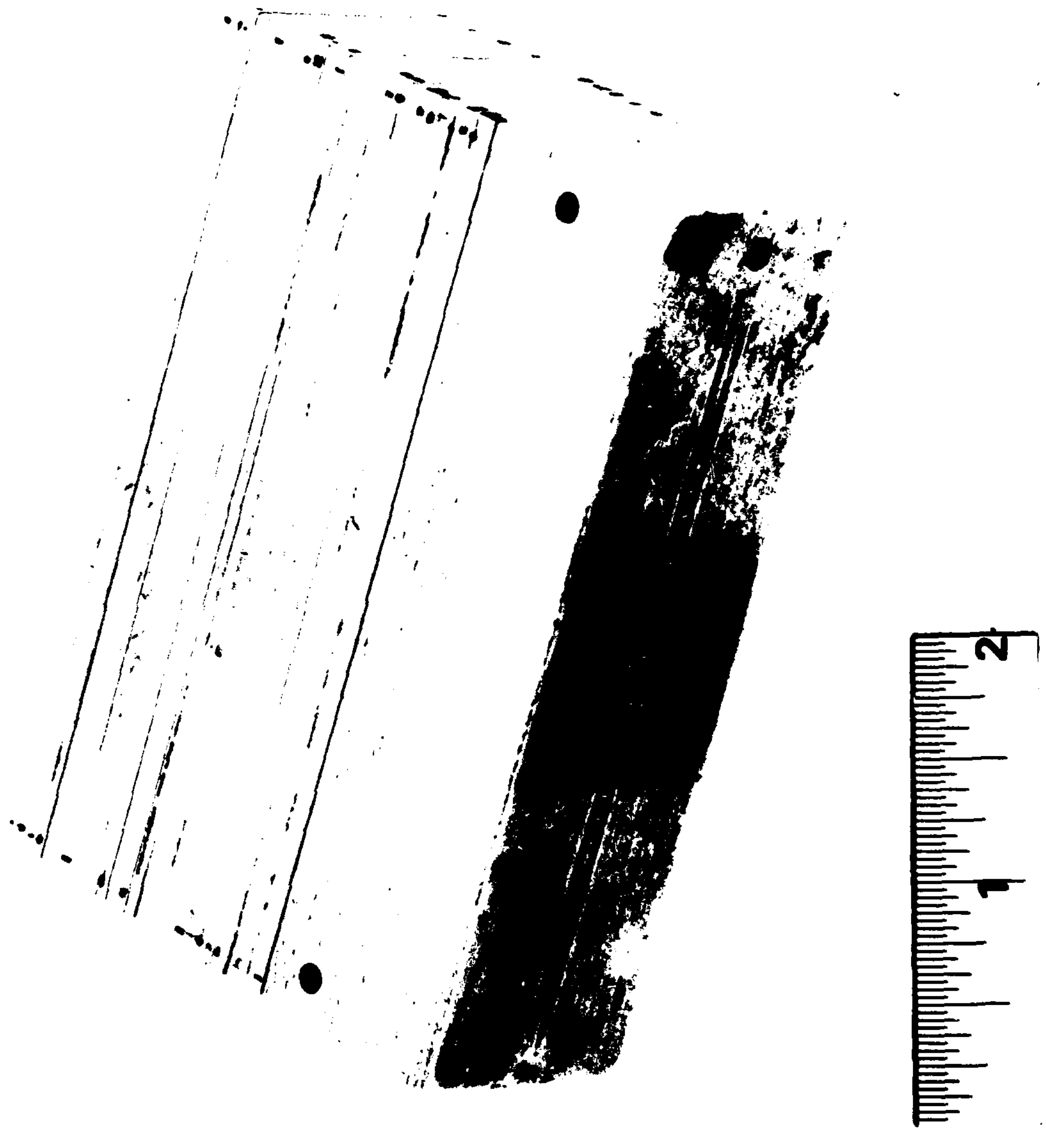

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Figure 15. Bonded Wafers and Trailing Edge Block for the 43.4% Reduced Solidity Radial Wafer Airfoil

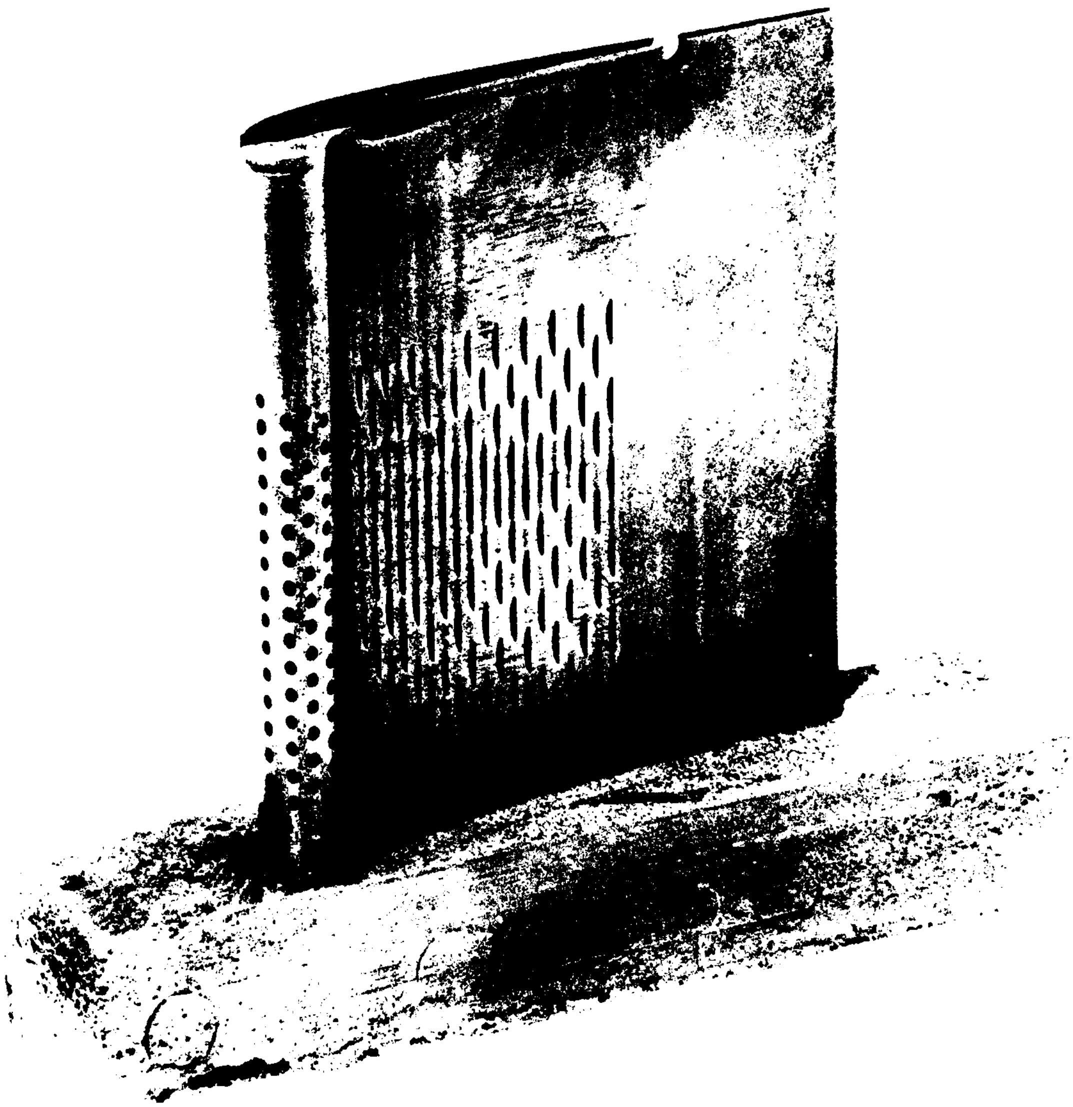

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Figure 16. 43.4% Reduced Solidity Radial Wafer Airfoil in Holder for Machining

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Figure 17. 43.4% Reduced Solidity Radial Wafer Airfoil

SECTION V

AIRFOIL CASCADE TEST

A. TEST RIG DESCRIPTION

The airfoil performance test was conducted in the plane cascade rig shown in Figures 18 and 19. Air is provided by compressor bleed from a J75 slave engine (Figure 20). The maximum airflow capacity is 28 lb/sec. Inlet airflow is controlled by a 10-in. control valve and measured with a sharp edged ASME standard orifice. Air temperature can be regulated from 160°F to 600°F with a water-cooled heat exchanger located upstream of the test rig.

Air is supplied to the test section through a 36-in. diameter plenum chamber, which is equipped with static pressure ports and temperature probes for determining rig inlet conditions. The rig has a rectangular transition duct or channel designed to provide two-dimensional flow conditions at the inlet to the test airfoil pack. Air inlet angle to the airfoil pack can be varied from -10 to 65 deg. The channel width in the airfoil gapwise direction can be varied up to a maximum of 18 in.; the channel height in the airfoil spanwise direction is 3 in.

B. TEST AIRFOIL DESCRIPTION

The cascade airfoil pack consisted of six constant cross section stainless steel slave airfoils and the radial wafer test airfoil located in the center position. Photographs of the assembled airfoil pack and a closeup of the test airfoil are presented in Figures 21 and 22 respectively. The test section passage height is 3-in., however, only the middle 1.65-in. was cooled. This height corresponds to the exit span of a typical first-stage turbine vane. Two air lines were utilized to supply the coolant to the test airfoil. One line supplied coolant to a common plenum which fed the leading edge, pressure side and trailing edge and the other line supplied coolant to the suction side radial passages.

The airfoil geometry selected for this program was the 43.4% reduced solidity turbine vane having potential application for an advanced Air Force fighter engine. The airfoil coordinates and pertinent design parameters for the reduced solidity airfoil are included in Table 2.

C. INSTRUMENTATION

In the plane cascade test, instrumentation was provided to measure airfoil pack total pressure losses, important inlet and exit main stream flow conditions, coolant inlet flow conditions, and airfoil wall temperatures. A schematic of the rig instrumentation is shown in Figure 23. The coolant inlet temperature and pressure to the test airfoil were determined for each of the supply lines. The airfoil was instrumented with 17 0.010 in. thermocouples embedded in the airfoil surface to measure wall temperature. The coolant and main stream flowrates were determined by using ASME calibrated orifices. The main gas stream inlet total temperature and pressure were measured by a traverse probe located at midspan which traveled from airfoil position 2 to 6. Exit total and static pressure and exit air angle were measured with a traverse probe located 0.2 in. downstream of the cascade. The exit probe traveled from airfoil position 2 to 6 at approximately 25% span. This location provided airfoil loss data that was not influenced either by the cascade endwall, or by the thermocouple slots which ran from midspan to tip.

All data were recorded on an automatic data recording system. Manually recorded data were obtained for the critical parameters also. Exit probe traverse and recording rates were selected to assure sufficient response time for servo-balance systems to accurately measure parameters, such as air angles and total pressures. All transducers measuring transients were close coupled to the sensor. Transducers used to measure critical parameters, such as cascade total pressure loss, were selected to have accuracies of ±0.2% of full-scale range. The smallest transducer ranges compatible with expected pressure levels were selected.

D. METHOD OF DATA ANALYSIS

1. Aerodynamic Performance

Airfoil aerodynamic efficiency (η) is evaluated by determining how nearly an isentropic process the available pressure energy is converted to velocity energy. In other words, it is defined as the ratio of the actual to ideal enthalpy drop as illustrated in Figure 24. The aerodynamic performance is usually expressed as the profile loss coefficient ($1-\phi^2$) which is defined as unity minus the efficiency. The profile loss coefficient may also be expressed in terms of upstream total pressure and downstream total and static pressure by:

$$1-\phi^2 = \frac{1-\left(\frac{P_{o1}}{P_{o2}}\right)^{\frac{\gamma-1}{\gamma}}}{1-\left(\frac{P_{o1}}{P_2}\right)^{\frac{\gamma-1}{\gamma}}}$$

where

$1-\phi^2$ = profile loss coefficient
P_{o1} = upstream total pressure
P_{o2} = downstream total pressure
P_2 = downstream static pressure
γ = specific heat ratio

2. Cooling Performance

The cooling performance is determined by measuring the airfoil wall temperatures at the design mainstream and coolant conditions. The cascade test was conducted at reduced temperature and pressure to reduce the complexity and cost of testing. The engine Reynold's number, Mach number and mainstream-to-coolant temperature ratio are duplicated at the reduced cascade test conditions for ideal heat transfer simulations. Testing at the reduced temperature, however, necessitates scaling-up of the wall temperatures to engine conditions to compare the data to the design prediction. The scaling is accomplished by using the cascade rig cooling effectiveness parameter (ϕ) defined as:

$$\phi = \frac{T_G - Tw}{T_G - Tc}$$

where

ϕ = cooling effectiveness parameter
T_G = gas stream temperature
Tw = wall temperature
Tc = coolant temperature

and replacing the gas stream and coolant temperature with their corresponding engine design values and solving for the engine wall temperature. This scaling procedure has been verified by comparing our reduced temperature rig results with results of high temperature cascade tests (Reference 1), as well as by NASA (Reference 2).

The average and hot spot cooling effectiveness parameters are defined as above, only Tw is replaced by the average wall temperature and the maximum wall temperature respectively. The two effectiveness parameters are then used to compare the cooling efficiency of various cooling schemes to one another.

The cold flow parameter ($W\sqrt{T}/PA$) is also used in determining the cooling capabilities of turbine airfoils. The function of the flow parameter is to establish the flow characteristics of the airfoil prior to testing. It is determined by varying the coolant supply pressure over a range from one to approximately two atmospheres and measuring the corresponding flowrate. The test is usually conducted with the coolant temperature and discharge pressure being equal to ambient conditions.

E. CASCADE AIRFOIL TEST RESULTS

1. Airfoil Flow Calibration

Before testing the airfoil, each section (leading edge, pressure side, suction side and trailing edge) was evaluated to determine the cold flow parameter ($W\sqrt{T}/PA$) versus pressure ratio. Since the leading edge showerhead, pressure side film slots, and the trailing edge wavy crisscross slot were all fed from one common supply, it was necessary to tape two sections while flowing the third. The suction side convective passages were fed from a separate supply and were evaluated independently.

The results from the cold flow study are presented in Figures 25 and 26. The test data agreed well with the predicted curve used during the design of the cascade airfoil with the exception of the pressure side film slots. The problem with flowing that section was that the leading edge showerhead holes were so concentrated and located so close to the first row of film slots that a good seal on the tape was difficult to obtain and the tape leaked. The pressure side data, although known to be in error, are included for completeness. The actual data are believed to be slightly higher than the design line as explained later.

The inspected area for the leading edge showerhead holes was obtained by measuring with calibrated gage wires. The results indicated an approximate 20% increase in coolant flow area above the design value. For the pressure side film slots, suction side convective passages, and the trailing edge wavy crisscross slot, the inspected area was obtained from selectively measuring approximately 30 times size photographs of RTV impressions of the passages obtained prior to bonding. A sample RTV photograph is presented in Figure 27. For the pressure and suction sides, the inspected area agrees well with the design area. In the trailing edge, the inspected area is approximately 13% oversized.

2. Aerodynamic Performance

The inlet total pressure and the exit total and static pressure across the cascade were obtained from the inlet and exit traverse probes respectively and presented as the aerodynamic loss coefficient ($1-\phi^2$). Figure 28 shows the uncooled and cooled aerodynamic performance of the conventional and the first cooled 43.4% reduced solidity airfoils measured in the plane cascade rig. Uncooled aerodynamic performance of the reduced solidity airfoil compared favorably with that of the conventional airfoil; however, the aerodynamic loss of the first cooled 43.4% reduced solidity airfoil was much higher. The primary cause for the increase in loss was the location of the suction side film cooling in a region of high surface Mach number. The cooling geometries for the two designs both having suction side film cooling are shown in Figure 29.

The cooling scheme for the redesigned 43.4% reduced solidity airfoil using the radial wafer fabrication technique is presented again in Figure 30. The film cooling on the suction side was eliminated with the use of the suction side radial coolant passages and the wavy crisscross slot in the trailing edge region. The aerodynamic performance obtained during the cascade evaluation of the airfoil is added to the previous data and presented in Figure 31. By eliminating the suction side film cooling, the aerodynamic loss for the 43.4% reduced solidity radial wafer airfoil was less than that obtained with the conventional airfoil and exceeded the program goal by 30%.

The profile loss coefficient for the test airfoil is presented again in Figure 32 as a function of Mach number. The data for the uncooled 43.4% reduced solidity airfoil and the data from the cooled conventional solidity airfoil are included in the figure. As indicated by the data, the cooled 43.4% reduced solidity radial wafer airfoil loss coefficient was slightly (17%) higher than the uncooled data and considerably (30%) lower than that obtained with the cooled conventional solidity airfoil at the design exit Mach number. The major reason for the improved aerodynamic performance is the elimination of the suction side film cooling. In the past airfoil designs when the suction side film cooling was in operation, the loss increased appreciably as shown in Figure 33 for the first cooled 43.4% reduced solidity airfoil. A similar plot (Figure 34) for the radial wafer airfoil at varying pressure ratios indicates no significant increase in loss with the suction side flowing.

3. Cooling Performance

Although the cascade rig conditions did not simulate the engine Reynolds number or temperature ratio, the cooling data do reflect the cooling characteristics of the design and indicate the same cooling trends in the trailing edge section as obtained in a separate trailing edge cooling study reported in Reference 3. A 10% improvement in cooling effectiveness based on the hot spot metal temperature was obtained with the radial wafer airfoil when compared to current vane data as illustrated in Figure 35. A plot of the wall temperature profile for the test airfoil at the cascade rig test conditions is shown in Figure 36. Also included on the figure are the scaled-up metal temperatures and the final design prediction. As shown by the figure, the agreement between the predicted and experimental data is good. In the suction side trailing edge section, the wavy crisscross design was actually more efficient than expected, and the measured wall temperature is approximately 100°F cooler than predicted. In the region of the suction side convective holes the experimental data agreed well with the prediction.

At the coolant design pressure ratio, the airfoil was overflowing approximately 20%. Roughly 6.7% and 3.7% of this increase can be accounted for with the increased flow area measured in the leading and trailing edge sections, respectively. The remaining increase in coolant is then attributed to the pressure side holes since the cold flow characteristics of that section were not defined due to tape leakage. This discrepancy in flow attributes somewhat to the difference between the predicted and measured wall temperatures.

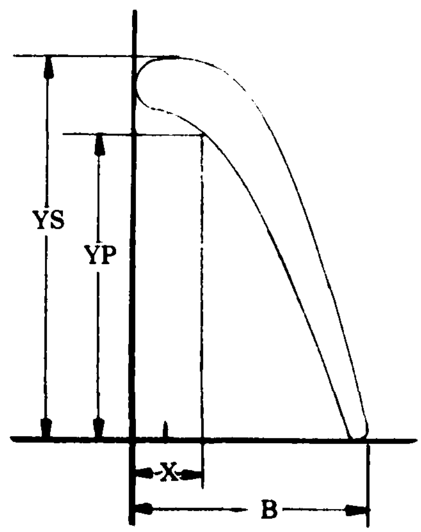
4. Cascade Test Summary

The cascade evaluation confirmed the fact that the utilization of efficient convective cooling designs on the suction side and in the trailing edge region of turbine airfoils will eliminate the need for film cooling and improve the aerodynamic performance relative to designs with suction side film cooling. The test, therefore, verified both parts to the goal of the program—to demonstrate efficient aerodynamic performance with improved convective cooling features that negate film cooling.

Table 2. Airfoil Description for 43.4% Reduced Solidity Airfoil

X/B	*X*	*YS*	*YP*
0.0000	0.0000	2.2735	2.2735
0.0100	0.0141	2.3392	2.2078
0.0200	0.0282	2.3642	2.1828
0.0300	0.0423	2.3819	2.1651
0.0400	0.0564	2.3954	2.1516
0.0500	0.0705	2.4061	2.1409
0.1000	0.1410	2.4358	2.1146
0.1500	0.2114	2.4457	2.0928
0.2000	0.2819	2.4367	2.0349
0.2500	0.3524	2.4061	1.9552
0.3000	0.4229	2.3546	1.8603
0.3500	0.4934	2.2815	1.7539
0.4000	0.5638	2.1872	1.6382
0.4500	0.6343	2.0731	1.5147
0.5000	0.7048	1.9408	1.3844
0.5500	0.7753	1.7924	1.2481
0.6000	0.8458	1.6304	1.1063
0.6500	0.9162	1.4569	0.9596
0.7000	0.9867	1.2739	0.8082
0.7500	1.0572	1.0830	0.6524
0.8000	1.1277	0.8858	0.4923
0.8500	1.1982	0.6833	0.3282
0.9000	1.2686	0.4763	0.1601
0.9500	1.3391	0.2658	0.0079
0.9800	1.3814	0.1380	0.0033
0.9900	1.3955	0.0951	0.0123
1.0000	1.4096	0.0450	0.0450

LER	0.1600
TER	0.0450
INLET ANGLE (GAS)	90.000
EXIT ANGLE (GAS)	20.1600
PITCH	2.6704
EXIT MACH NO.	0.6920
NUMBER OF FOILS	26.000



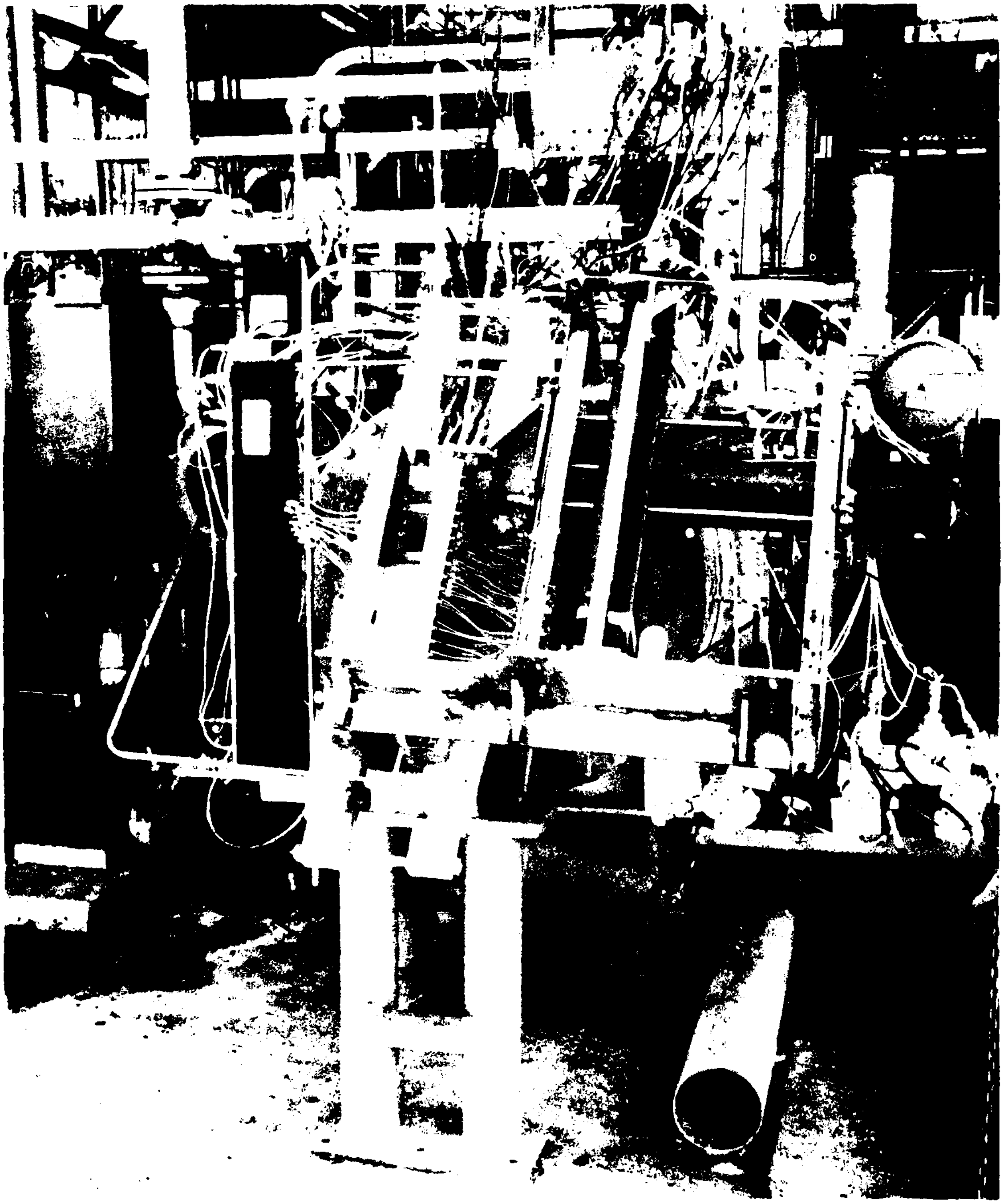


Figure 18. Plane Cascade Rig

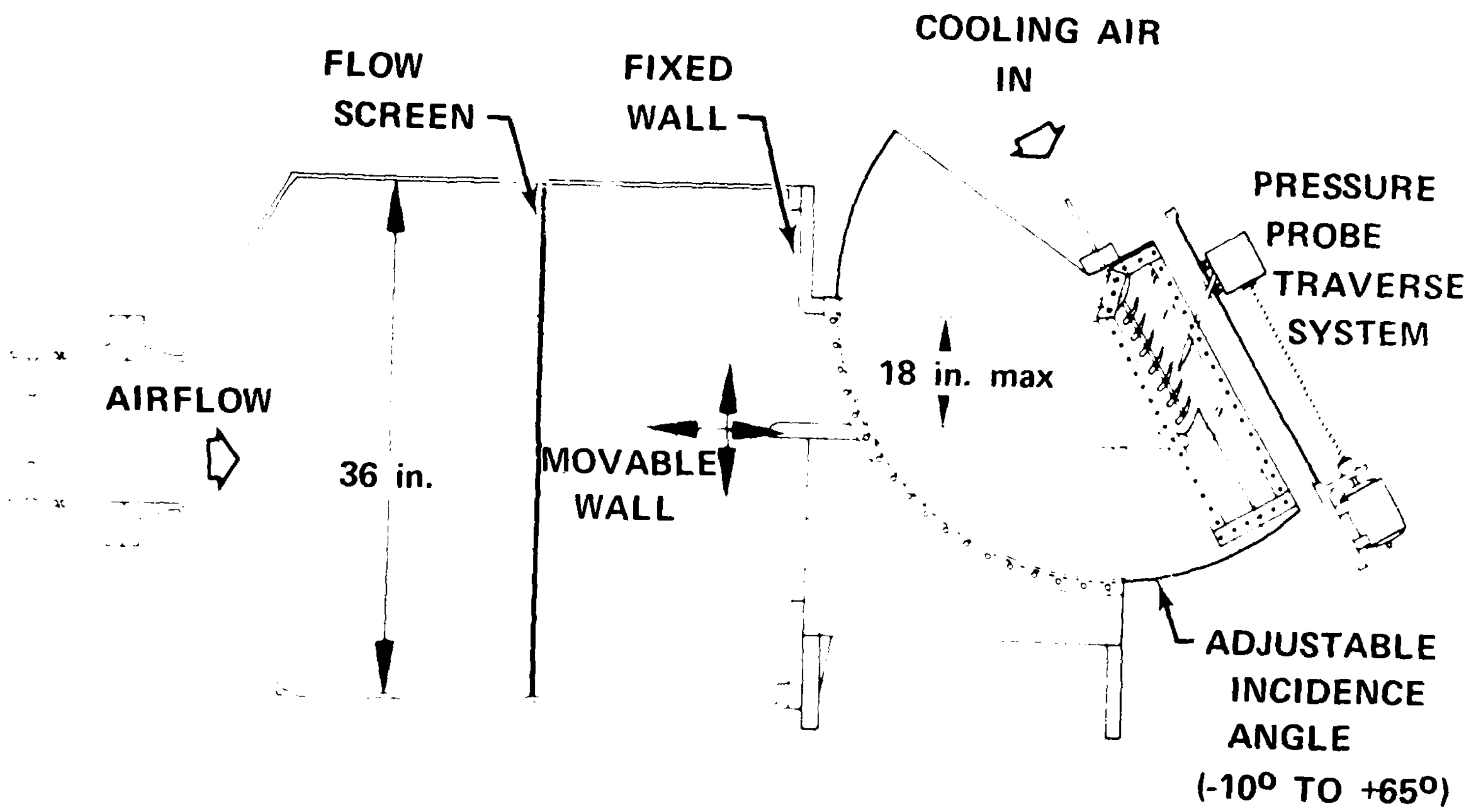


Figure 19. Turbine Plane Cascade Rig

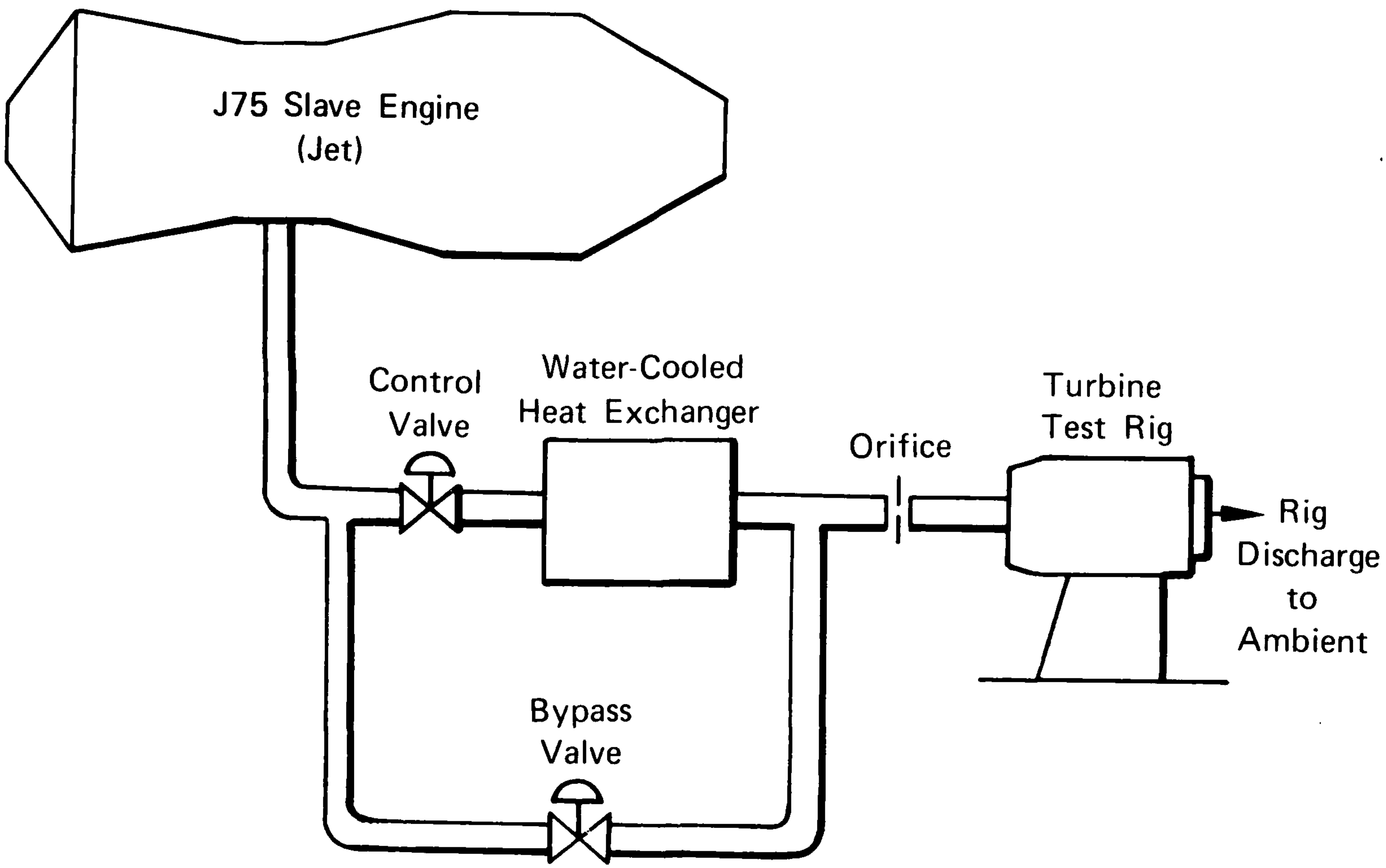


Figure 20. Aerodynamic Turbine Rig Test Facility

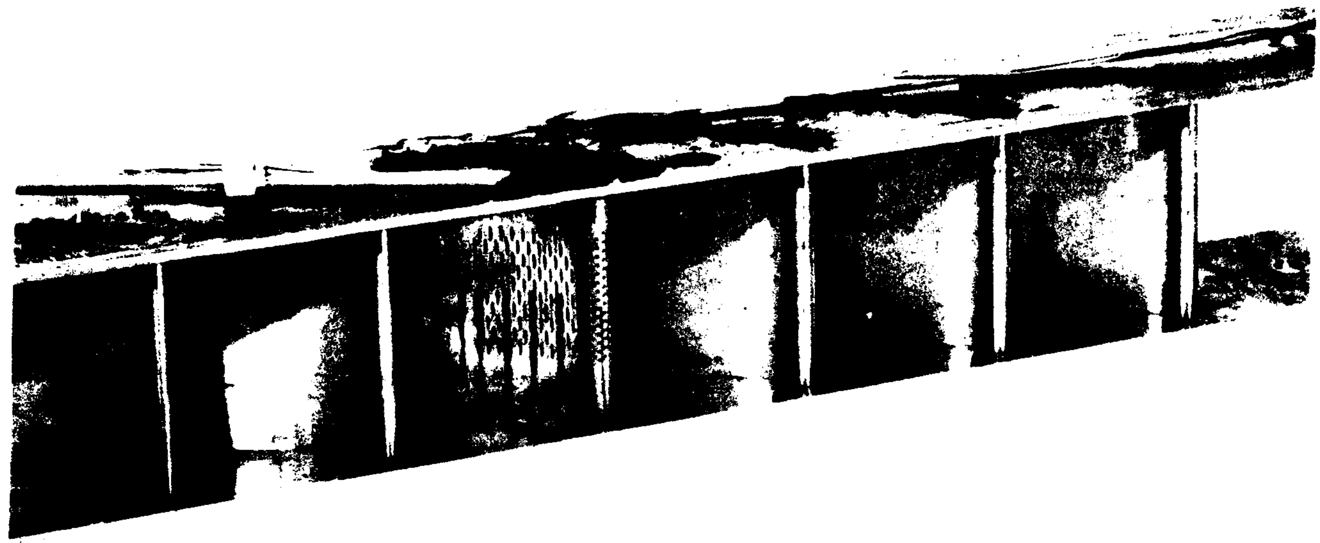


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Figure 21. Cascade Assembly for the 43.4% Reduced Solidity Radial Wafer Airfoil

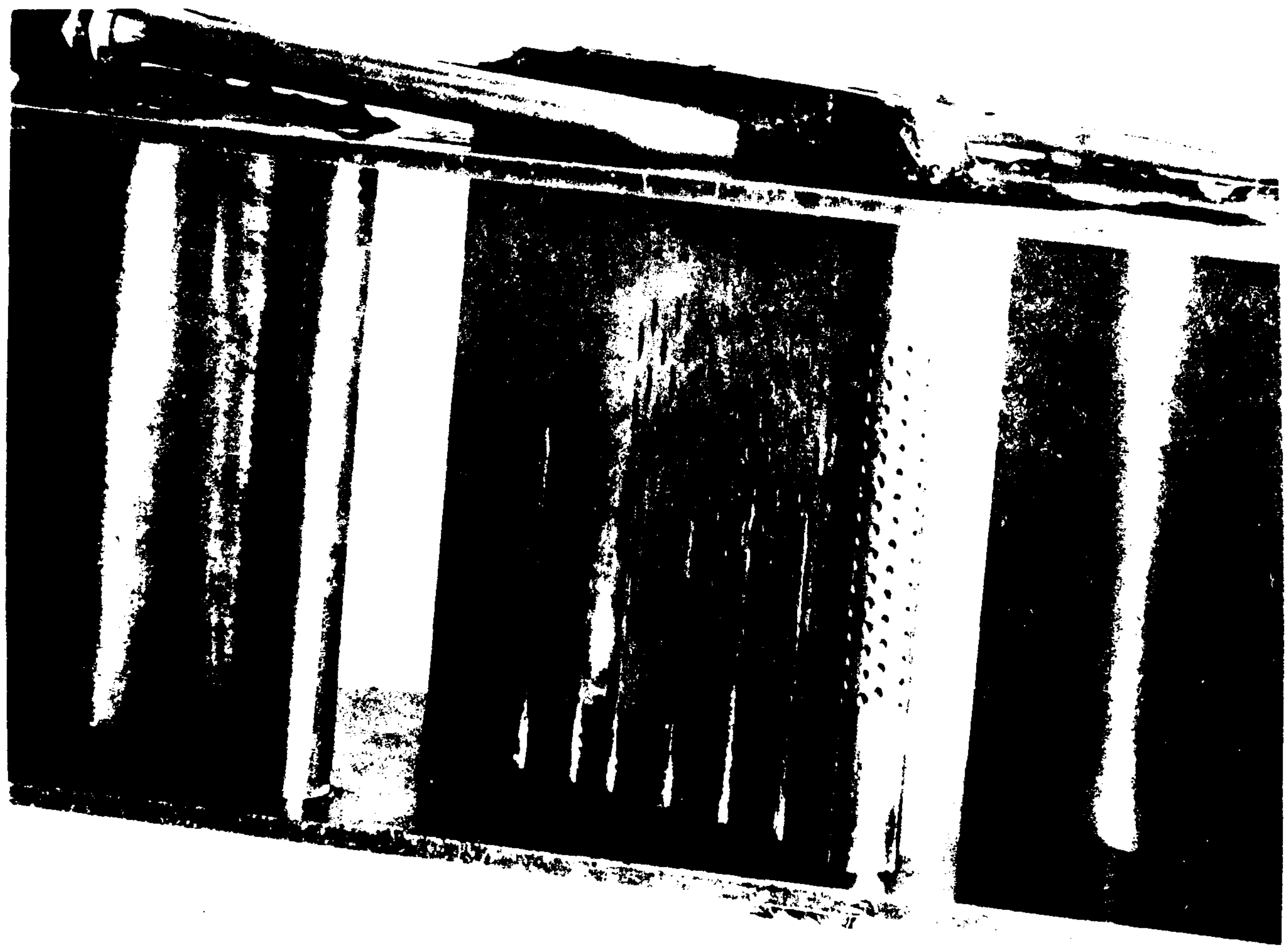

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Figure 22. Close-up of the 43.4% Reduced Solidity Radial Wafer Airfoil Within Cascade Assembly

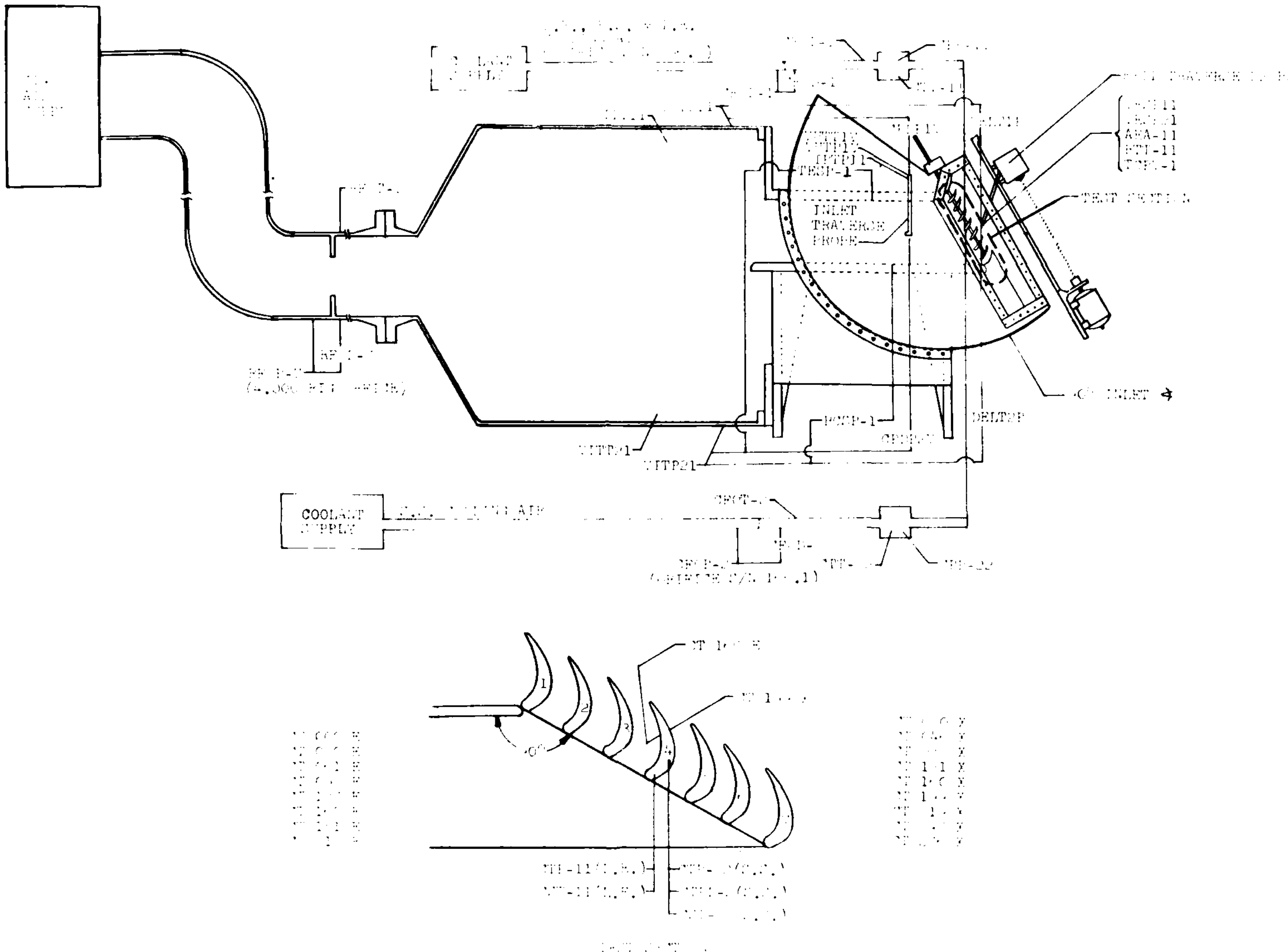

Figure 23. Cascade Rig Instrumentation Schematic for the 43.4% Reduced Solidity Radial Wafer Airfoil

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$$\eta = \frac{ho_1 - h_2}{ho_1 - h'_2} = \frac{V_2^2}{V'^2_2}$$

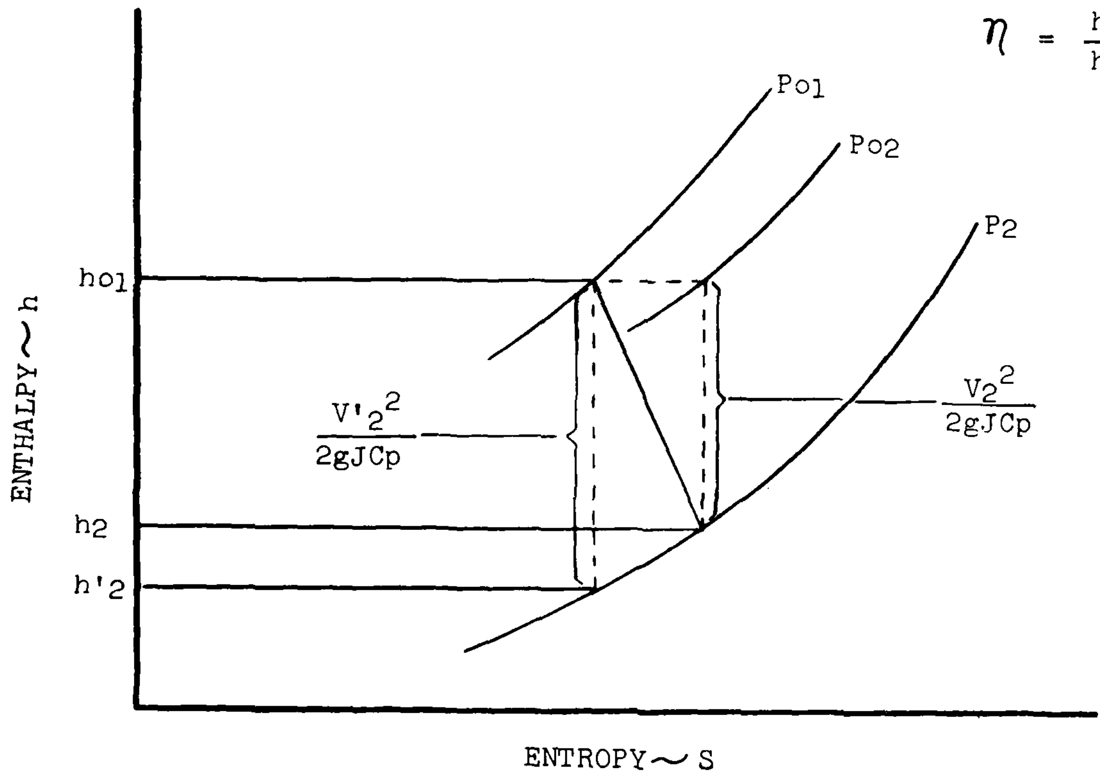
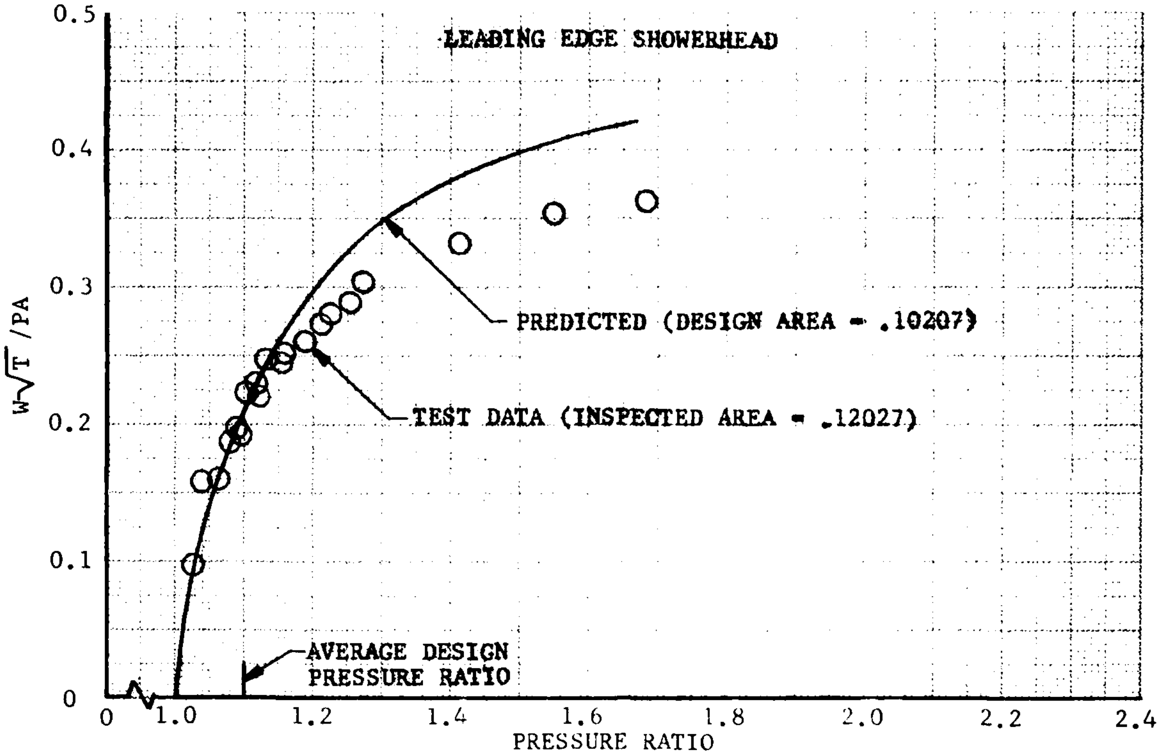


Figure 24. Definition of Airfoil Aerodynamic Efficiency



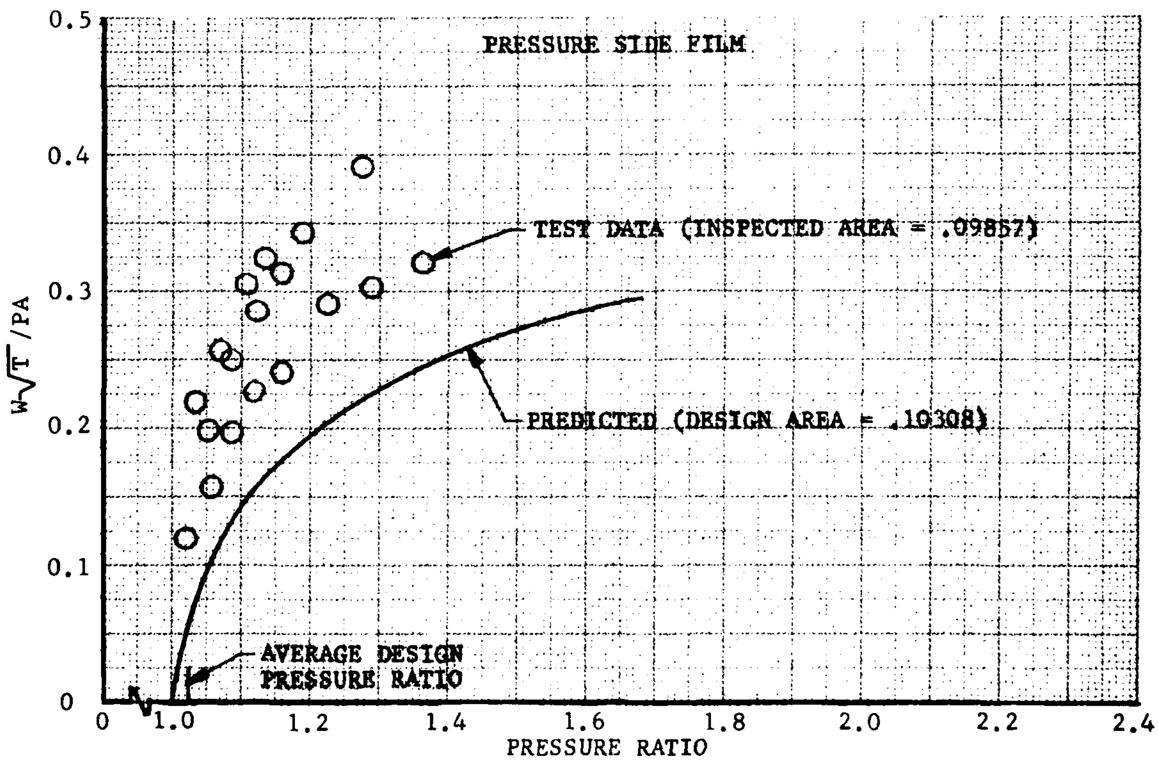


Figure 25. Flow Results for the Advanced Cooled Turbine Airfoil

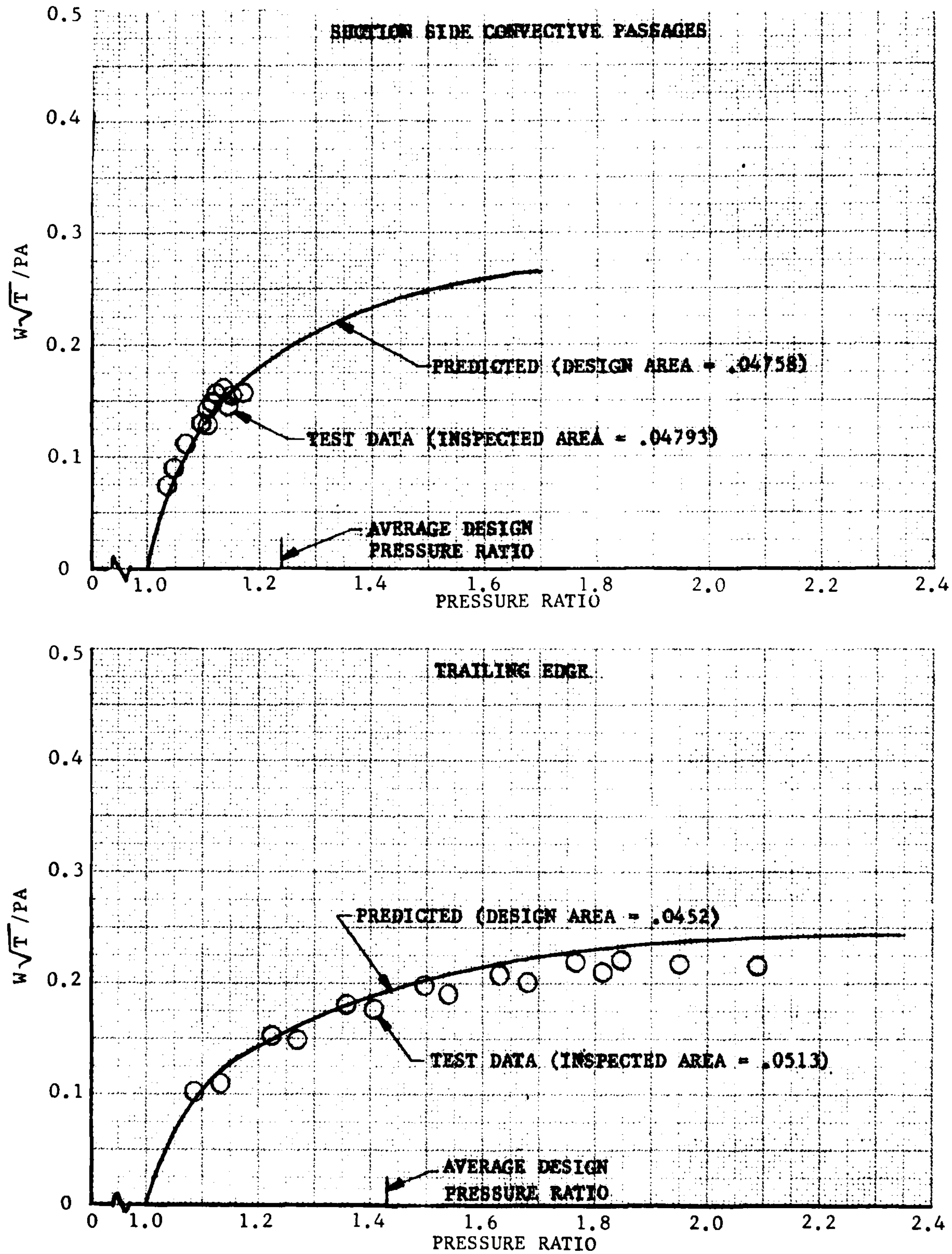


Figure 26. Flow Results for the Advanced Cooled Turbine Airfoil

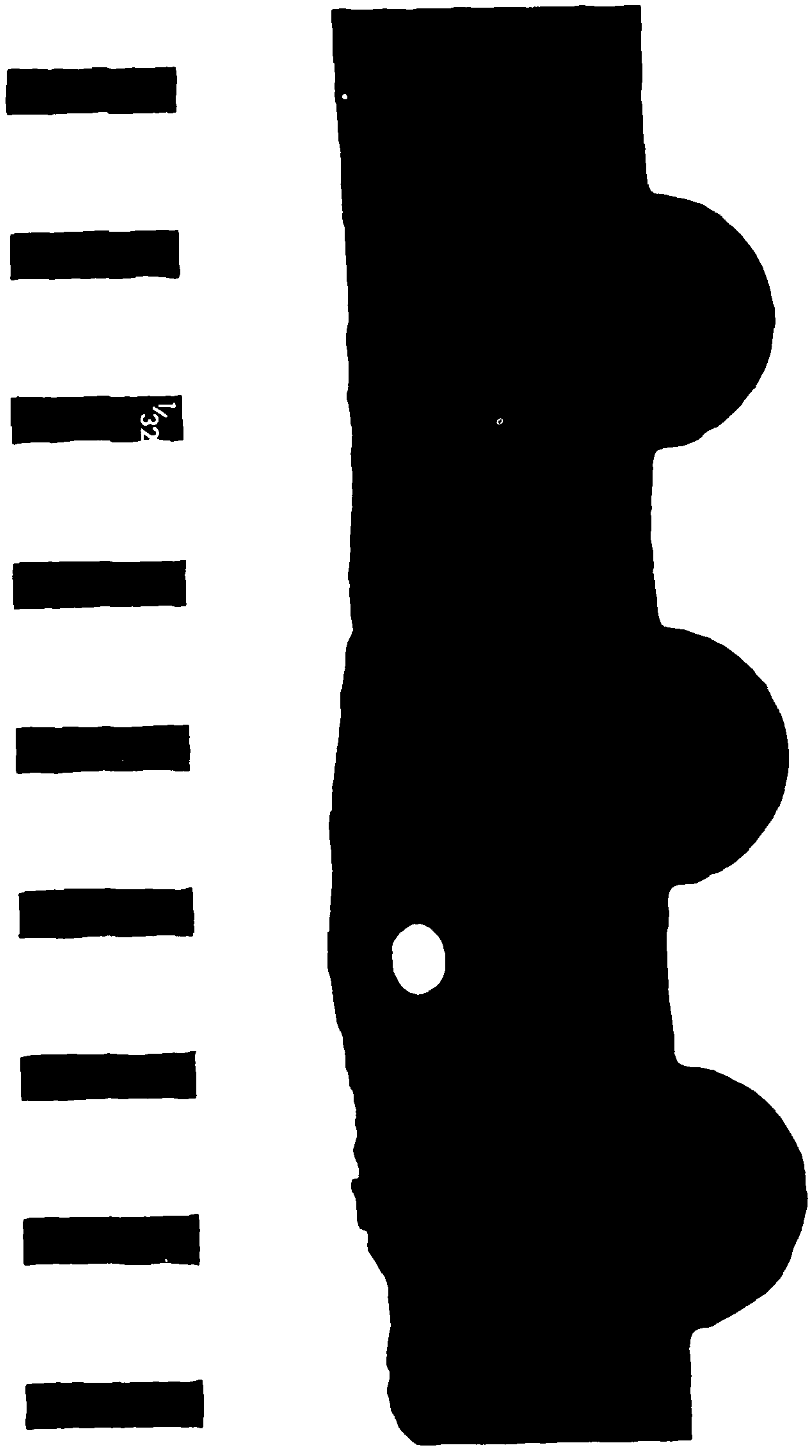


Figure 27. RTV Impression of Wafer No. 10 Film Slot

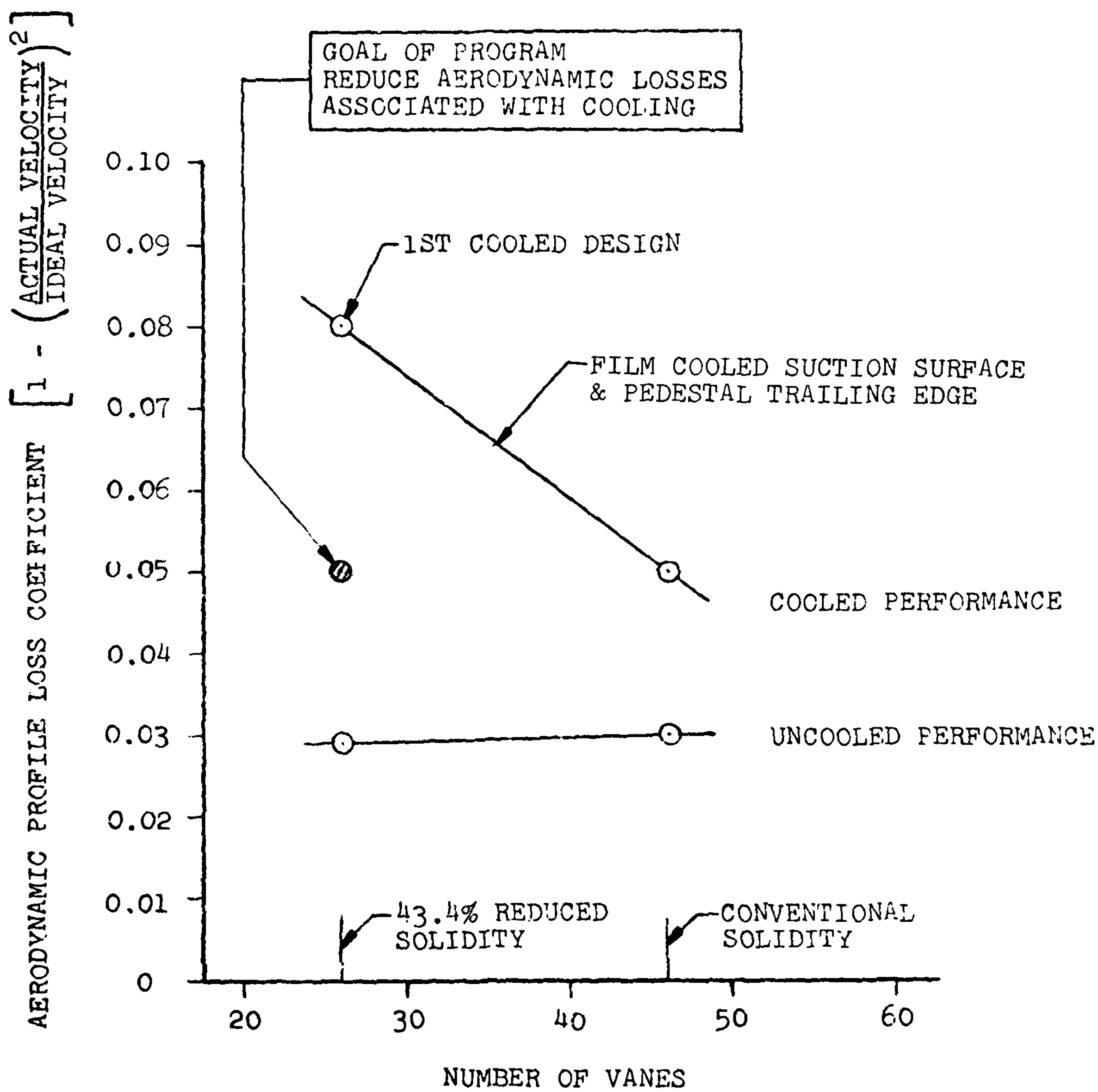


Figure 28. Aerodynamic Performance for the First Vane Airfoils

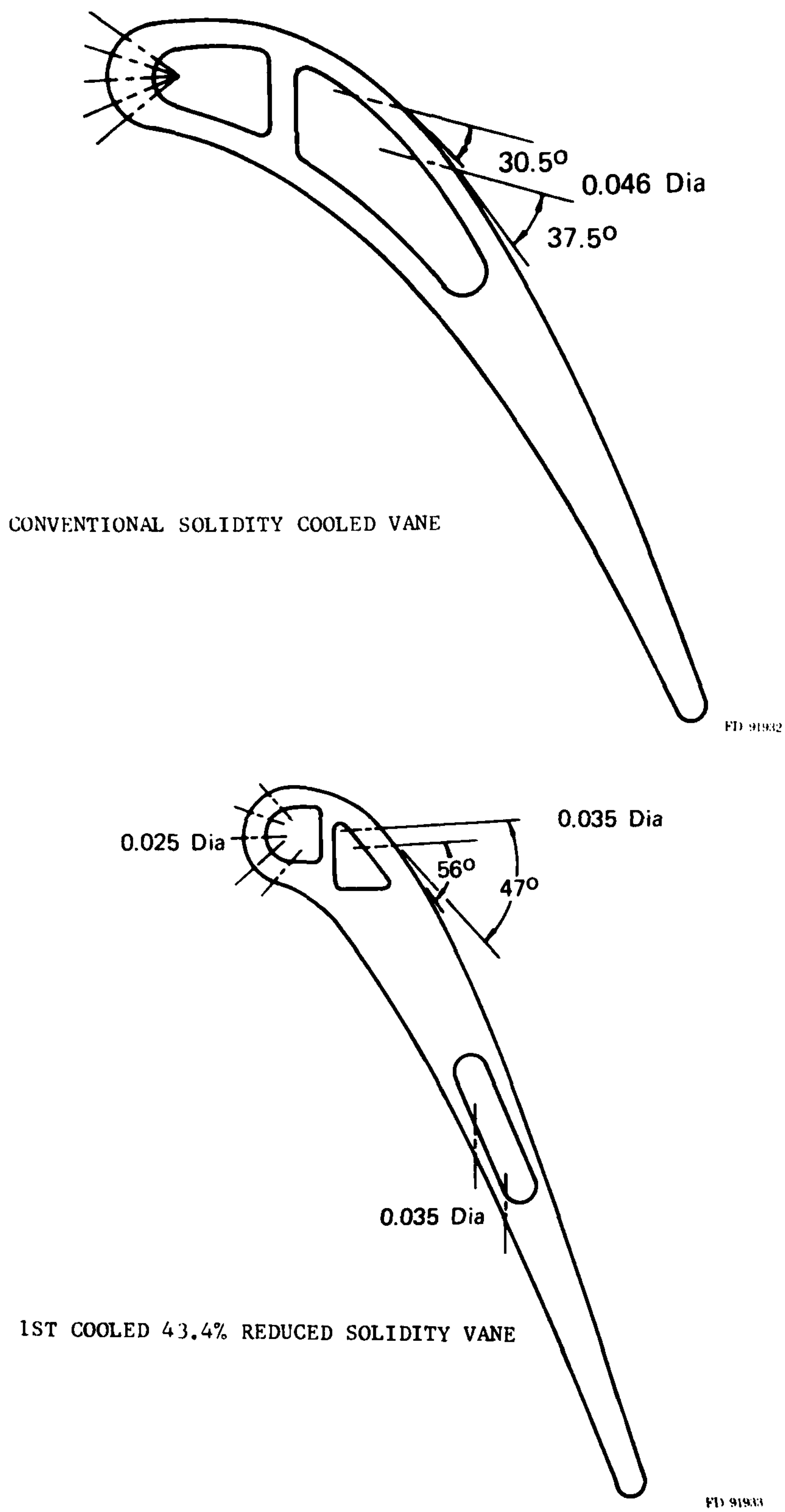


Figure 29. Cooling Configurations for First Vane Airfoils

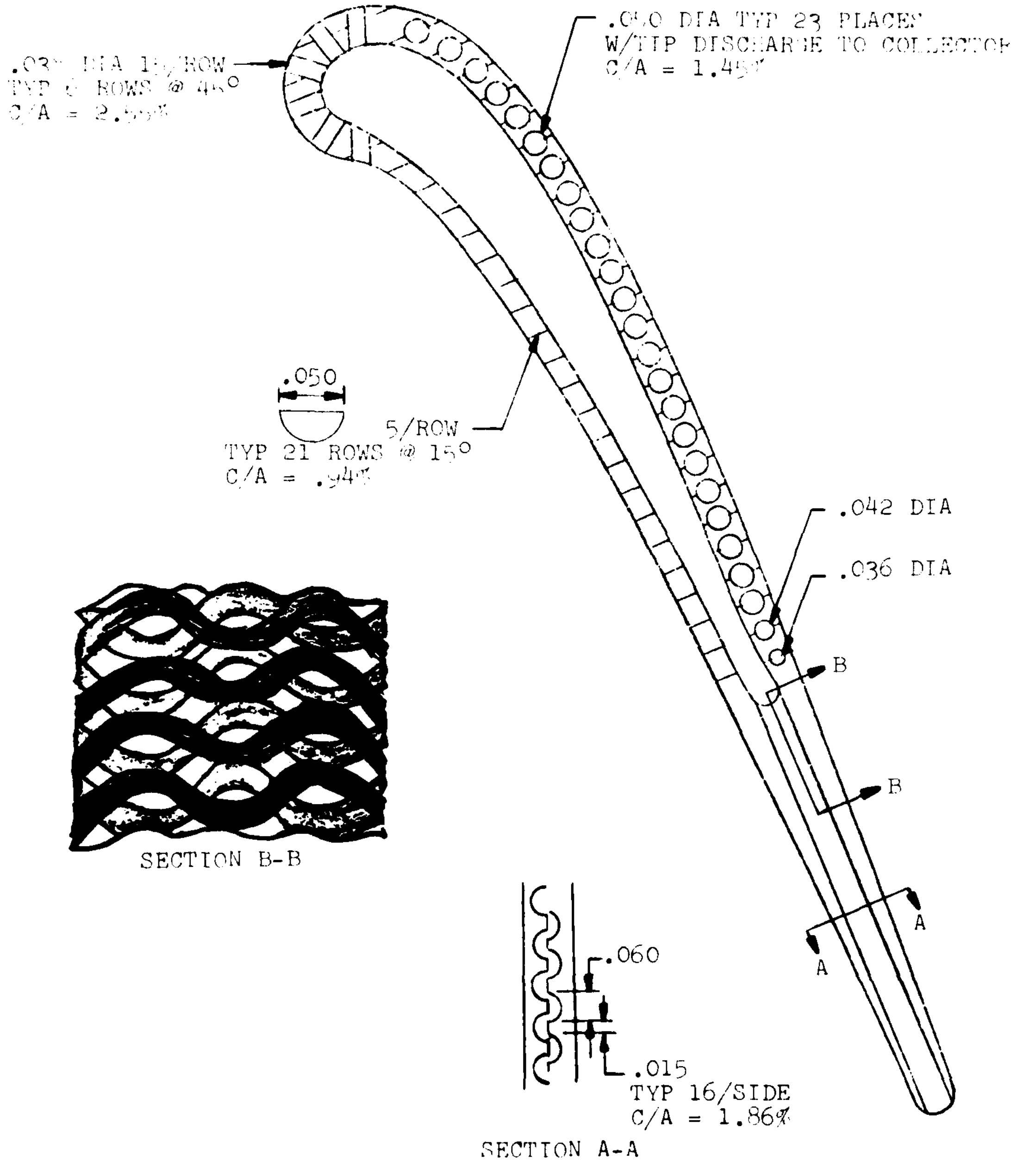


Figure 30. 43.4% Reduced Solidity Radial Wafer Airfoil - Final Design

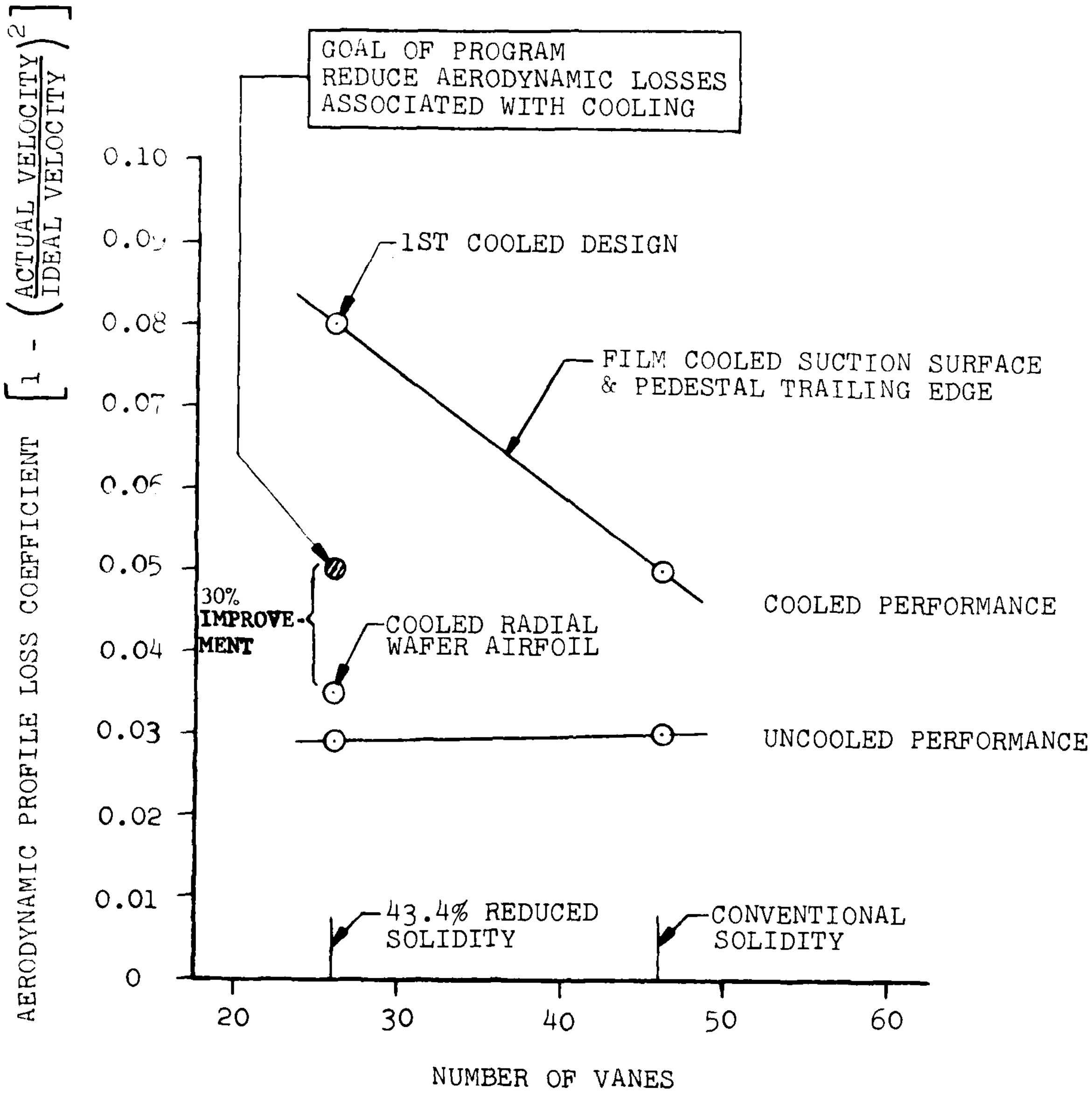


Figure 31. Aerodynamic Performance for First Vane Airfoils

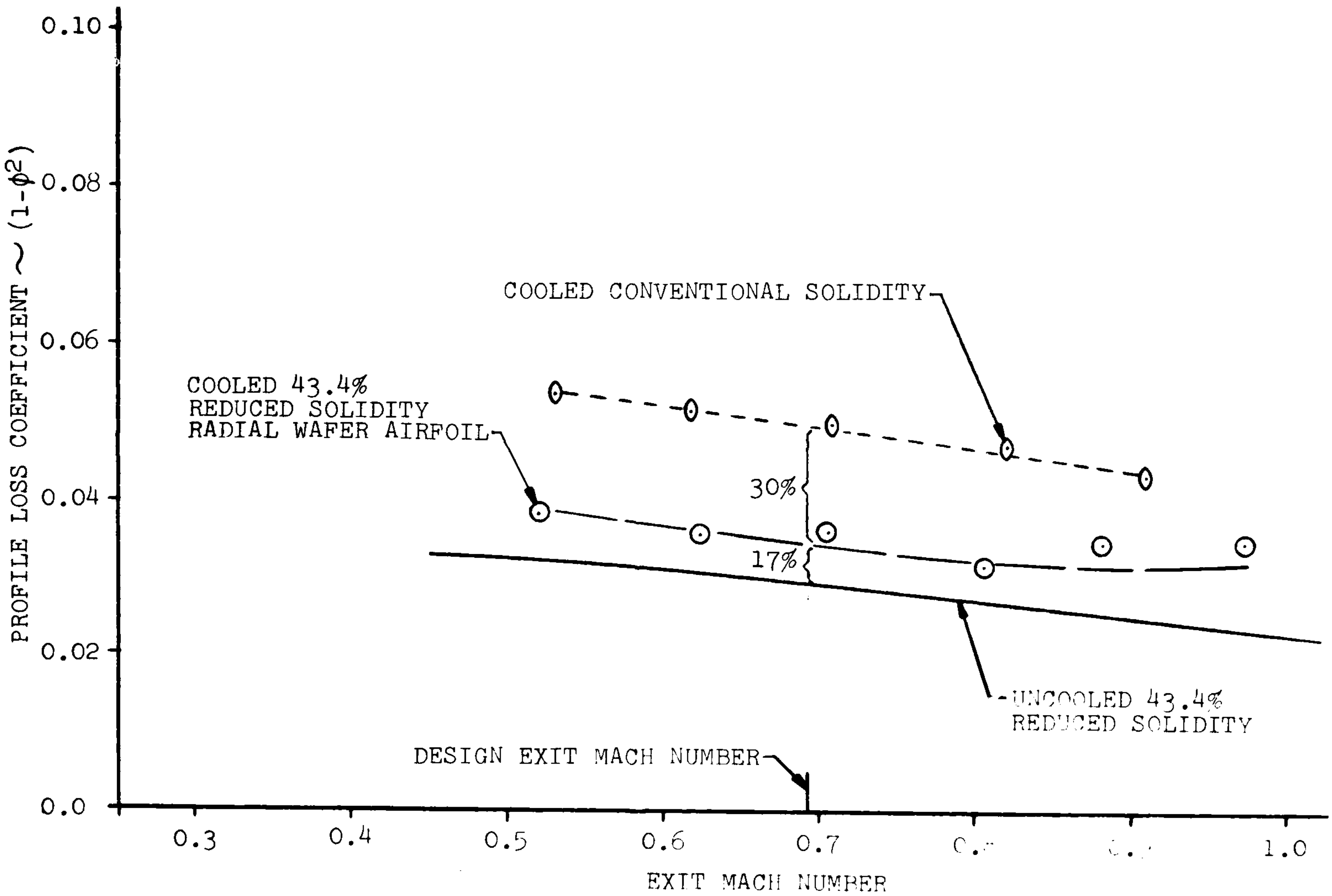


Figure 32. Profile Loss Coefficient Versus Exit Mach Number for First Vane Airfoils

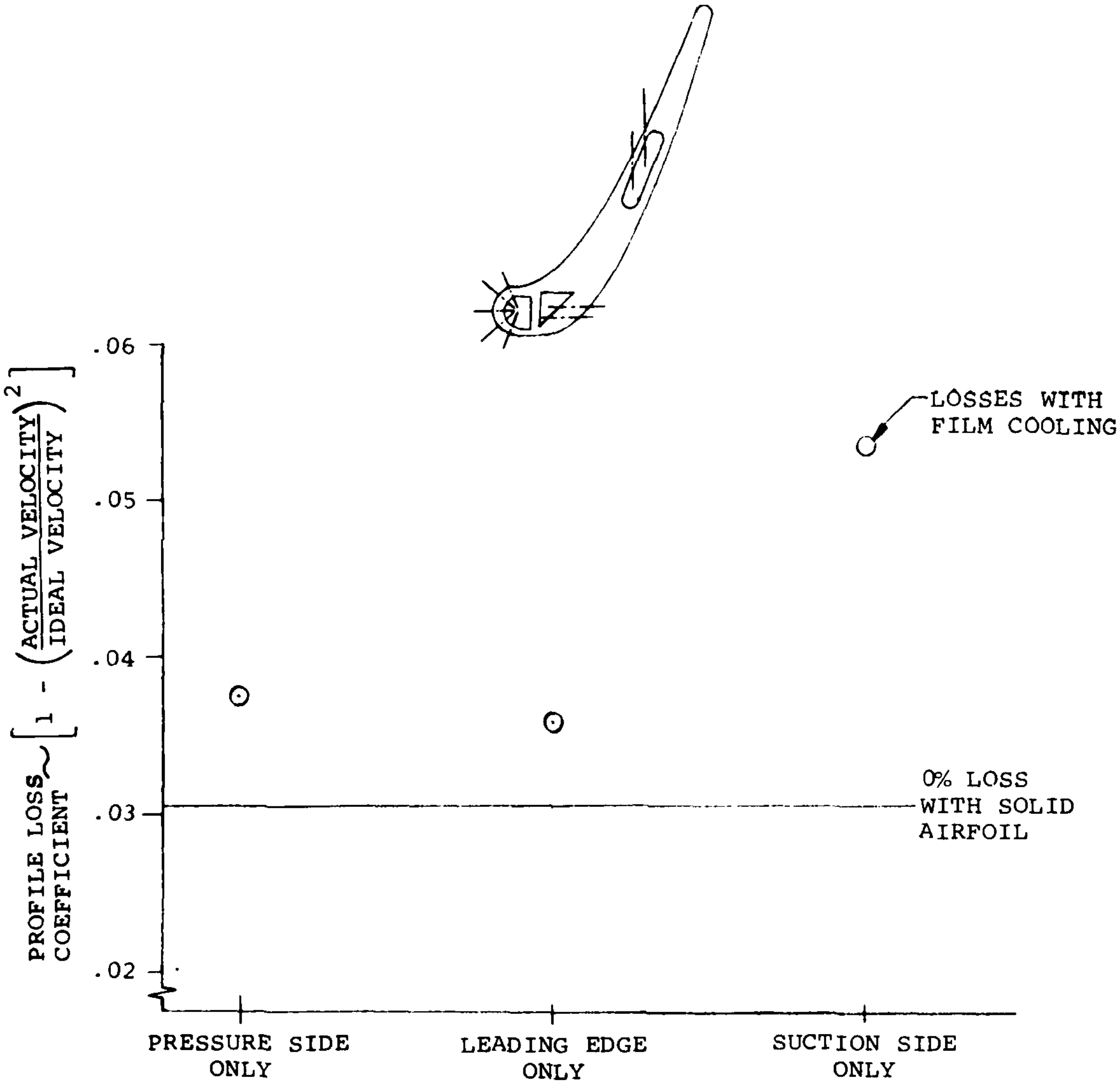


Figure 33. Profile Loss Coefficient for the First 43.4% Cooled Reduced Solidity Vane

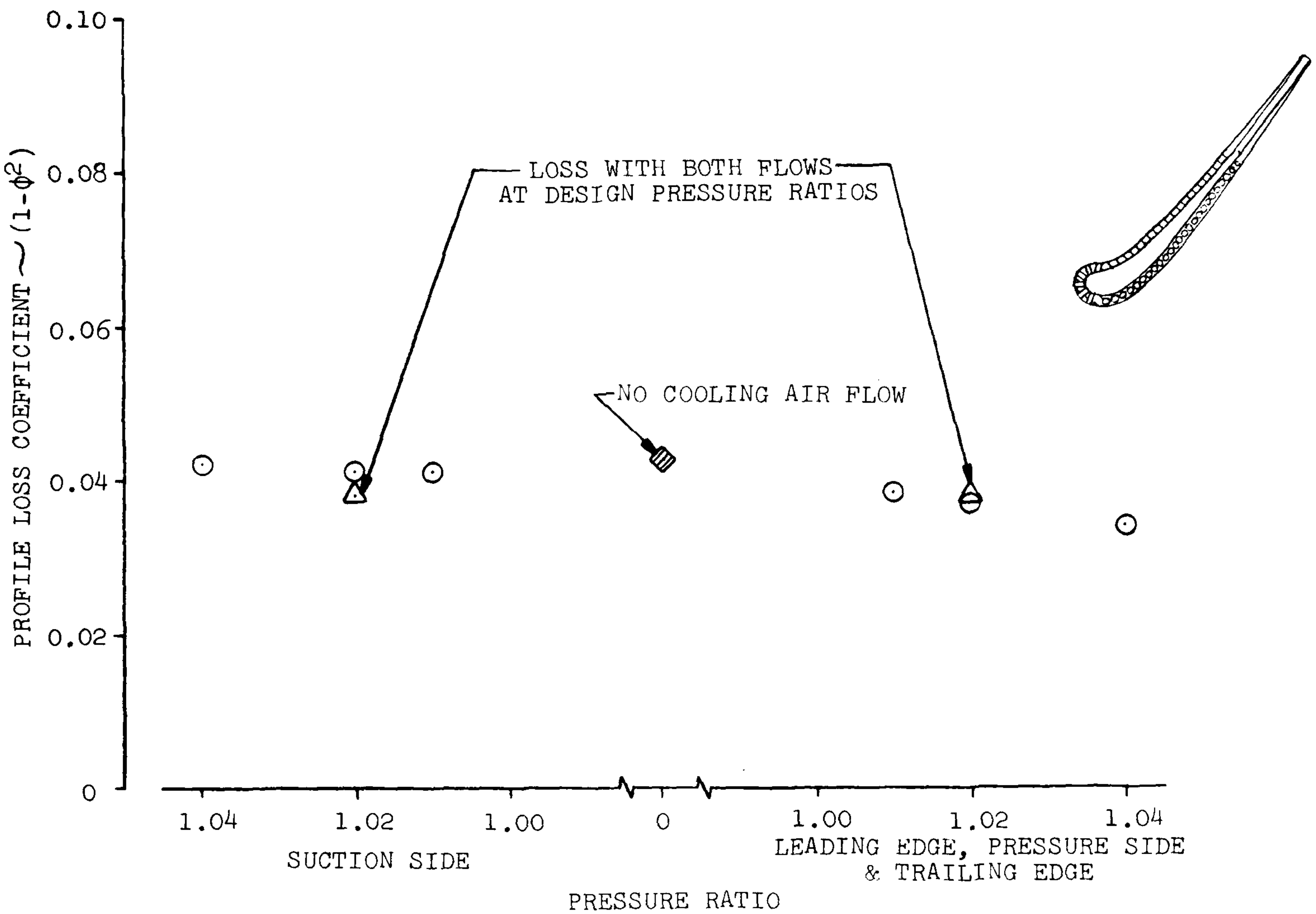


Figure 34. Profile Loss Coefficient Versus Pressure Ratio for the 43.4% Reduced Solidity Radial Wafer Airfoil

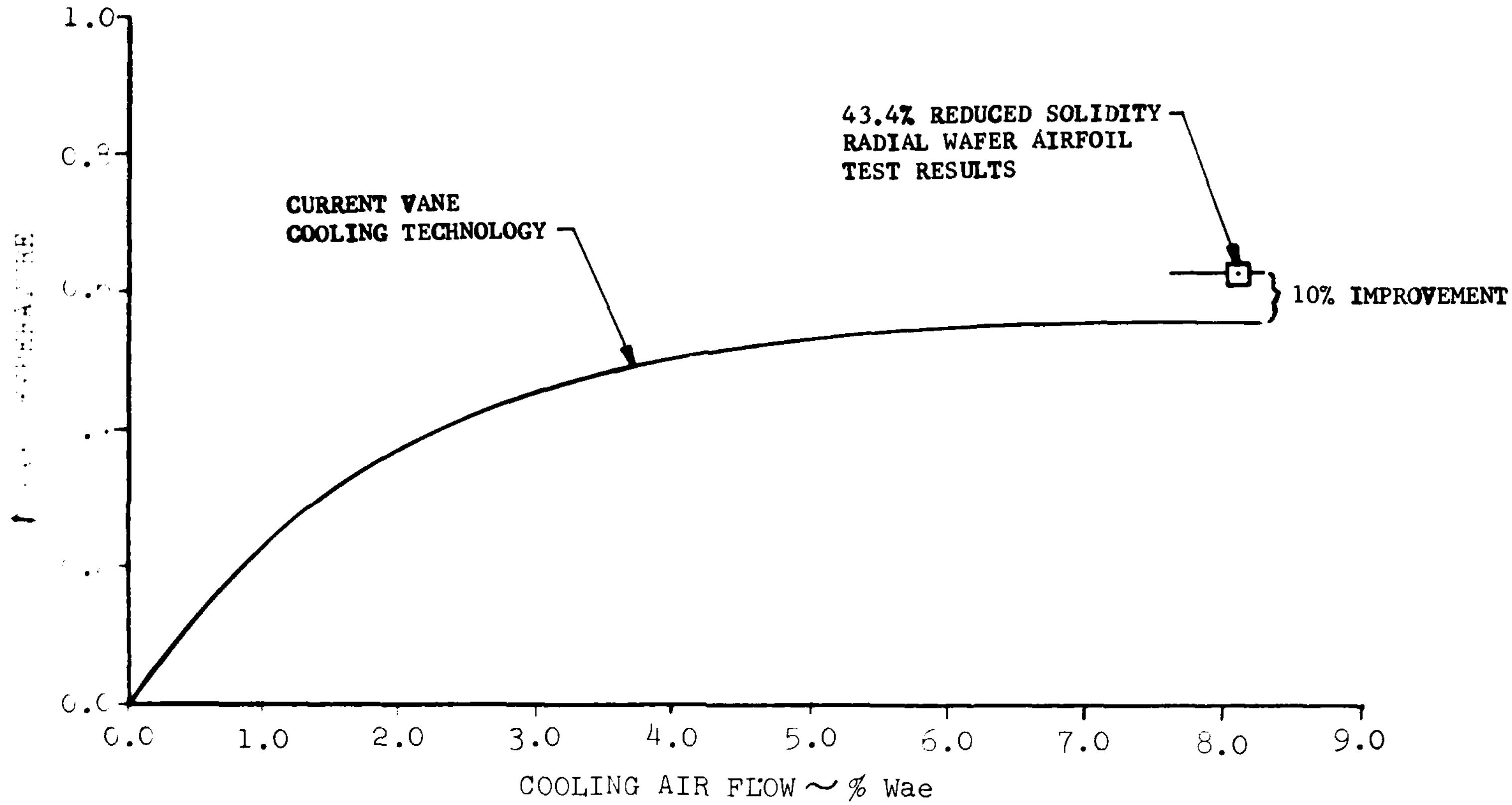


Figure 35. Vane Cooling Effectiveness

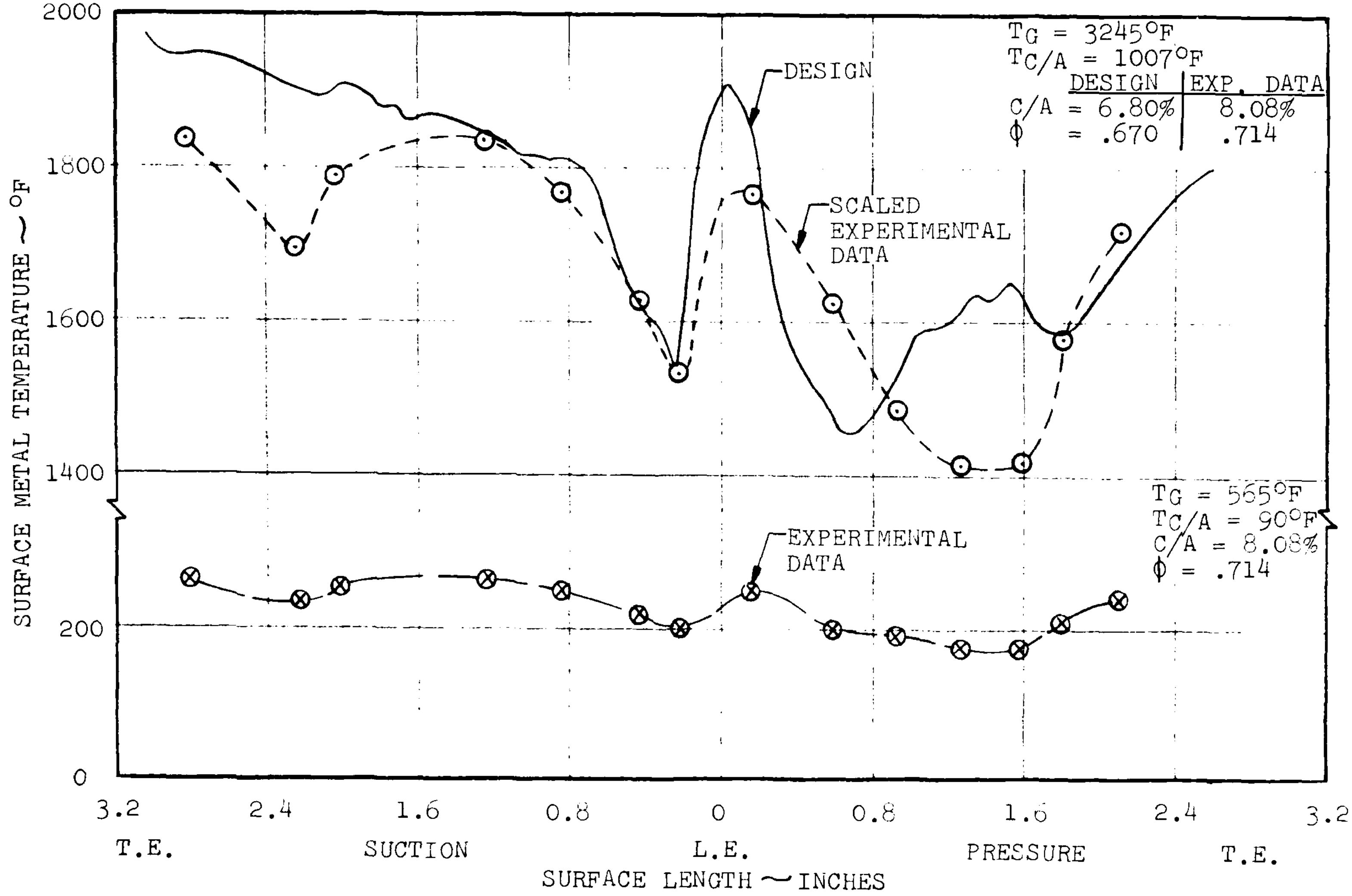


Figure 36. Temperature Profile for the 43.4% Reduced Solidity Radial Wafer Airfoil

SECTION VI
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1. Independent Research and Development Program, 1973 Pratt & Whitney Aircraft, FP 72-104, November 1972.

2. Gladden, H. J., and Livingood, J. N. B., "Procedure For Scaling of Experimental Vane Airfoil Temperatures from Low to High Gas Temperature," NASA TN D-6510, September 1971.

3. Hess, W. G. to Mitchell, J. P., "Summary Report for the Trailing Edge Model Tests," April 29, 1976.